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RAMJET TECHNOLOGY

Chapter 3

ENGINE REQUIREMENTS FOR SUPERSONIC FLIGHT

by

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Chapter 3

E N G I N E   R E Q U I R E M E N T S
F O R   S U P E R S O N I C   F L I G H T

by

James H. Walker

Applied Physics Laboratory
The Johns Hopkins University

(Manuscript submitted for publication
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3. ENGINE REQUIREMENTS FOR SUPERSONIC FLIGHT

by

James H. Walker

3.1 INTRODUCTION

The accelerated development of jet powerplants immediately prior to World War II and continuing to the present has sponsored an intense interest in the achievement of supersonic flight. To state that the powerplant is the staff upon which supersonic flight is wholly dependent is obviously an oversimplification, but it is, none the less, true that a transition from reciprocating engines to jet powerplants was an essential step to be taken before supersonic flight could become a reality.

It is not possible to argue persuasively that any one jet engine is the best possible engine for all supersonic applications. Each type has its shortcomings as well as its advantages. The selection of a particular engine thus is necessarily a function of a specific problem. A valid comparison necessarily avoids arbitrary constraints on the operation of different types of engines which are not significant to accomplishment of the basic mission; i.e., a rocket need not necessarily follow a ramjet trajectory nor use the same guidance system to accomplish the desired objective. Analogous situations are frequently encountered. Thus, it is evident that the development of a satisfactory criterion to judge the context must be accepted as an essential ingredient in studies of engine selection and optimization. Failure to recognize all characteristics of a particular situation (e.g., tactical) and boundary conditions (e.g., method of delivery) may result in failure of the optimization process to accomplish its real purpose.

This chapter does not attempt to cover every facet of jet engine application. Consideration of accelerating, except for ramjet boosters, and over-the-atmosphere rockets, is beyond the scope of the present effort. It is believed that the following discussion will, however, assist the reader who wishes to gain some insight into factors which influence application of the ramjet engine, and who requires knowledge of the special advantages of this engine for missile applications on a contemporary time scale.

3.2 FUNDAMENTAL PARAMETERS OF FLIGHT PERFORMANCE

It is possible for preliminary design and engine-comparison purposes to reduce a large number of supersonic flight variables to a relatively few terms which describe the airframe and powerplant. While these terms are not sufficiently accurate to allow application in their simplest form to an engineering design, they are valuable in outlining general regions of interest. Boundary-condition overlays may be used to increase the value of studies based on these parameters and are helpful in determining the proper direction in which to proceed to achieve maximum results. In many instances such overlays are also informative as to the nature of limitations which can weight the balance in optimization or selection procedures.

Lift-Drag Ratio

The lift and drag and lift-drag ratio are basic parameters in the analysis of propulsion requirements and fuel capacity for vehicles designed to fly atmospheric flight paths. This is obvious from consideration of the following expressions of equilibrium conditions required for nonaccelerated flight.

$$F = D = C_D \, q \, A \qquad (3.2\text{-}1)$$

$$W = L = C_L \, q \, A \qquad (3.2\text{-}2)$$

Typical values of the lift and drag coefficients for a supersonic configuration are given in Fig. 3.2-1. Lift coefficient curves are generally linear with angle of attack to fifteen or twenty degrees. Drag coefficients, on the other hand, demonstrate a nearly quadratic variation with angle of attack. Substitution of typical values in the lift equations will demonstrate that a very large aerodynamic lift may be generated by a very small angle of attack. In fact for sea-level, supersonic flight a fraction of one degree frequently provides adequate lift to support a typical missile. This fact is a consequence of the extremely large dynamic pressures associated with supersonic, sea-level flight. It should be noted in Fig. 3.2-1 that the associated drag for sea-level equilibrium flight may equal or exceed the value of lift.

Division of the first expression by the second demonstrates that the thrust requirement is inversely proportional to lift-drag ratio. It is obvious, therefore, that lift-drag ratio should be as high as possible to insure minimum engine weight and minimum fuel load for a specified objective.

Figure 3.2-2 demonstrates a typical variation of the ratio lift to drag. It is evident that a very large increase in the value of this term is possible if the configuration can be operated at several degrees angle of attack. Thus it follows, from the preceding paragraph, that equilibrium supersonic operation at the angle required for maximum lift-drag ratio is not possible at sea level. Either climbing flight must be accepted or the angle of attack reduced, with consequent deterioration of the lift-drag ratio.

Another alternative is to increase flight altitude to take advantage of reduced air density to increase angle of attack for equilibrium flight and thus improve the lift-drag ratio. The effect of altitude has been investigated for several configurations and Mach numbers and plotted in Fig. 3.2-3. It is notable that an order of magnitude improvement results from high-altitude operation as compared to sea-level operation. Limitations which operate to prevent flight at the required altitude for maximum lift-drag ratio will, it is obvious, serve

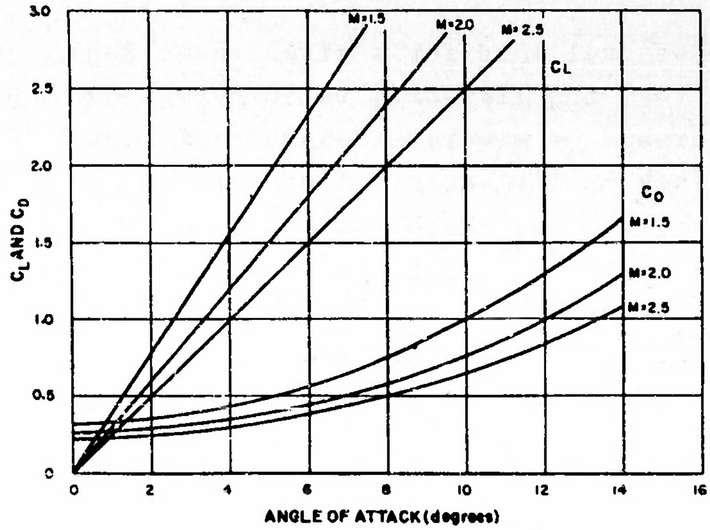

Fig. 3.2-1 LIFT COEFFICIENT (C_L) AND DRAG COEFFICIENT (C_D) AS A FUNCTION OF ANGLE OF ATTACK AND MACH NUMBER FOR A TYPICAL SUPERSONIC CONFIGURATION

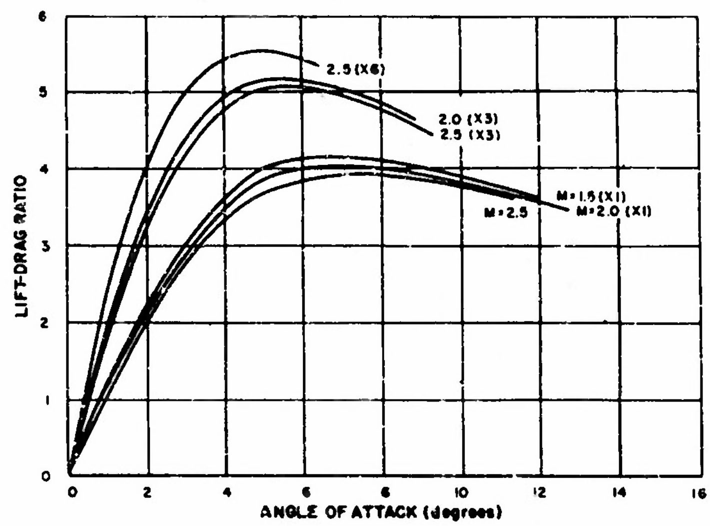

Fig. 3.2-2 LIFT-DRAG RATIO DEPENDENCE ON ANGLE OF ATTACK, MACH NUMBER, AND WING AREA FOR A TYPICAL SUPERSONIC CONFIGURATION

(x1) Basic Wing Area
(x3) Three Times Basic Wing Area
(x6) Six Times Basic Wing Area

to increase fuel load for a given range requirement. It should be noted that the lift-drag ratio curves do indicate an altitude tolerance of several thousands of feet with a small percentage loss in lift-drag ratio. If, however, the limiting altitude is very low, the optimum wing area in this instance will be reduced to vanishing proportions. The latter statement is made with reference to range maximization only and it is, of course, possible that wing area may be determined by other than range considerations (e.g., maneuverability requirements).

It has been implied, though not stated, that lift-drag ratio must be considered in terms of a maximum value for a physical configuration and an operating value dependent on flight altitude. Maximum lift-drag ratio is largely dependent on Mach number, on wing characteristics and on the fraction of total drag contributed by the body and control surfaces and associated forces which do not contribute to net lift. Estimated maximum obtainable lift-drag ratios for various types of wings with reasonable body configurations are shown in Fig. 3.2-4 [1,2,3].

The large reduction in lift-drag ratio which accompanies an increase in Mach number from subsonic to supersonic is notable and is most significant to required engine performance.

A considerable amount of aerodynamic research is being conducted to increase obtainable lift-drag ratios. It behooves the engine designer to keep abreast of these developments because of the effect of sizeable changes on required engine characteristics. One outstanding possibility is related to the possible existence of large areas of laminar flow on supersonic configurations which can be designed to operate at relatively high altitude and Mach number. This subject is in an exploratory stage at the moment, but indications are that proper choice of Mach number and flight altitude may increase the estimates of Fig. 3.2-4 markedly, with consequent benefits particularly to

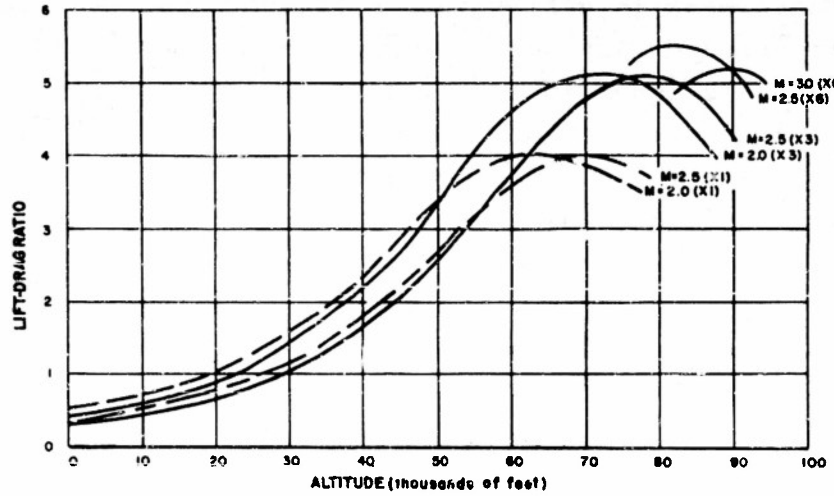

Fig. 3.2-3 LIFT-DRAG RATIO DEPENDENCE ON FLIGHT ALTITUDE, MACH NUMBER, AND WING AREA FOR A TYPICAL SUPERSONIC CONFIGURATION

(x1) Basic Wing Area
(x3) Three Times Basic Wing Area
(x6) Six Times Basic Wing Area

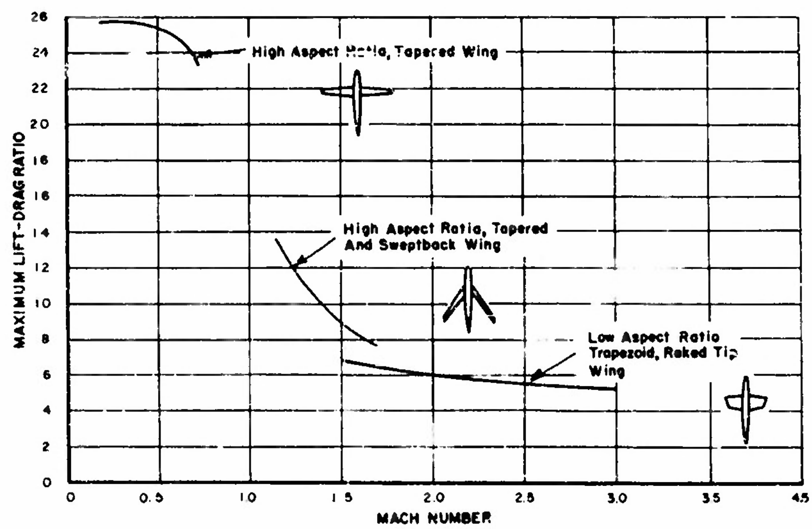

Fig. 3.2-4 ESTIMATED MAXIMUM LIFT-DRAG RATIOS FOR TYPICAL AERODYNAMIC CONFIGURATIONS AS A FUNCTION OF MACH NUMBER AND WING SELECTION

long-range missile performance. It must also be recognized that the present inadequate knowledge on this subject places the engine designer in a somewhat hazardous position because of the uncertainty associated with prediction of required thrust coefficients. The present tendency toward conservatism in drag estimates, which results from assumption of turbulent flow over supersonic configurations, may lead the engine designer to design engines which will not operate satisfactorily at lowered thrust coefficients associated with possible (but uncertain of prediction) laminar flow. Obviously, a cautious viewpoint and further study is indicated [4].

Thrust per Square Foot of Frontal Area

Powerplants are housed in aerodynamic bodies such as fuselages, nacelles, wings, etc. The drag of these bodies is proportional to their size and particularly to their cross-section area if bodies of average fineness are considered. Thus, it is evident that engines should develop large thrust per unit area (and volume) to minimize external drag. The criticalness of this parameter is related to the lift-drag ratio of the aerodynamic configuration and the extent to which engine dimensions determine configuration dimensions. In the case of low subsonic-speed aircraft, configuration lift-drag ratio and related thrust requirements are compatible with the thrust capabilities of reciprocating engines [see Fig. 3.2-5]. Thus, it is possible to utilize such engines efficiently at low speed. At even low supersonic speed, however, the attainable lift-drag ratio is so low, particularly when engine-induced compromises are considered, and the thrust obtainable per square foot of engine is so small that a match between engine and airframe is no longer possible. Fortunately, jet engines can assume responsibility for propulsion in the supersonic flight

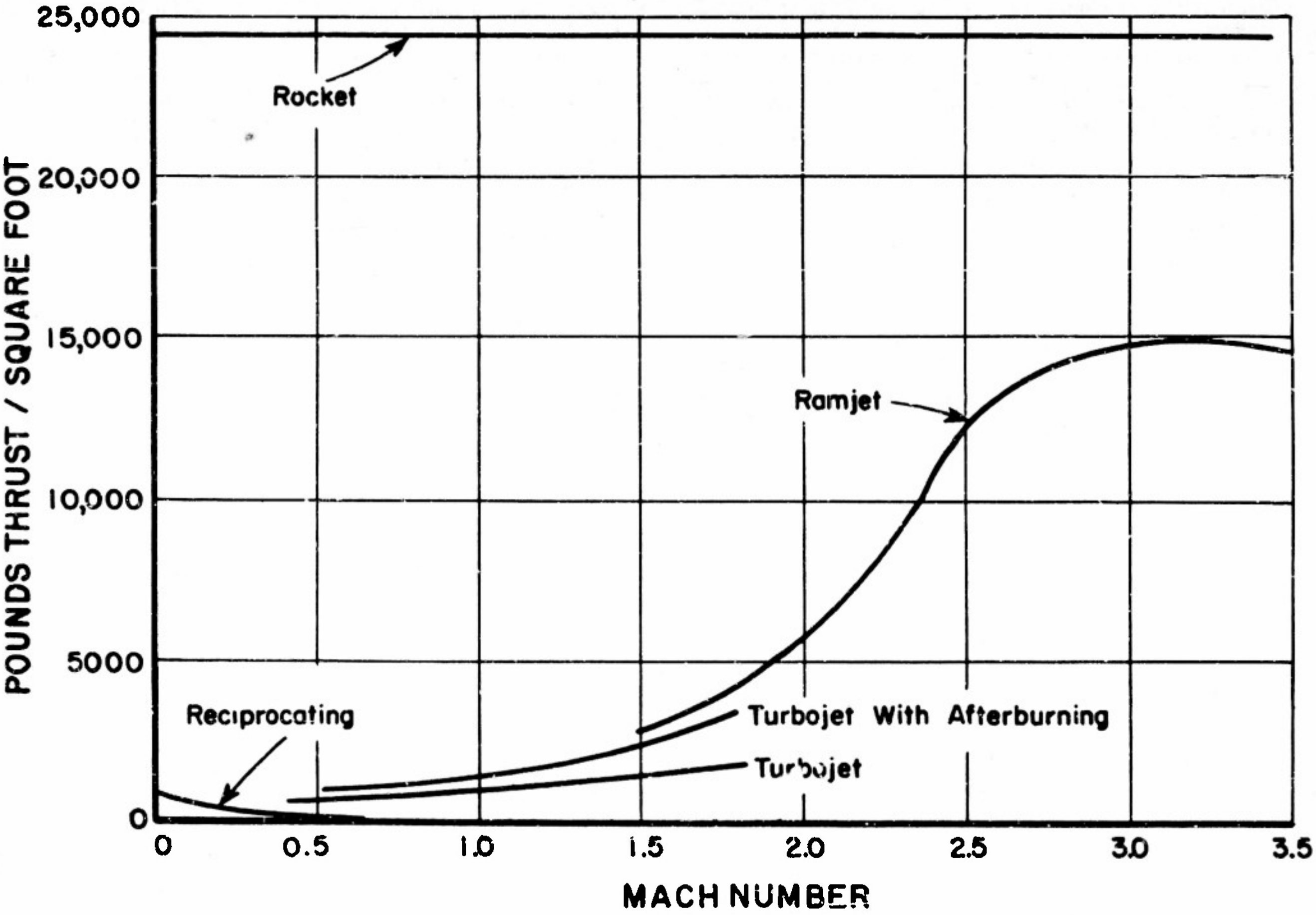

Fig. 3.2-5 TYPICAL THRUST AVAILABLE PER SQUARE FOOT OF FRONTAL AREA FOR ENGINES OF VARIOUS TYPES (SEA LEVEL OPERATION)

regime and can provide enormous thrust per unit area (and per unit volume) when compared to the reciprocating engine. Figure 3.2-5 illustrates the situation which exists for several powerplants over a large speed range.

The values given in Fig. 3.2-5 are representative only; in particular instances values may vary considerably by virtue of the limitations of a particular application. It is essential to realize that the integration of jet engine and airframe is a much more intimate and subtle wedding of components than is the case in reciprocating engine aircraft design [5,6,7,16]. Although less than ideal solutions will generally be realized in practical cases, the designer should strive to evaluate fully the interplay which exists between the engine and airframe. For example, both external drag and economy characteristics of jet engines can be favorably influenced through proper relation of engine geometry, packaging volume, body fineness ratio, and body diameter with consequent beneficial effects on the absolute fuel consumption rate and tankage required.

Thrust per Pound of Engine Weight

The engine weight required to produce a given thrust is a second factor of great significance in engine selection. Returning to the previous subsonic reciprocating engine example, it is evident that reasonably high engine weight to produce thrust can be tolerated if the lift-drag ratio is high. However, anticipated supersonic lift-drag ratios are far less than subsonic values, with a correspondingly greater penalty required to transport the engine weight of a supersonic vehicle. Table 3.2-1 has been prepared to illustrate the expected variation in thrust per unit weight for several engines. Inasmuch as supersonic flight is the concern of the present text, values are presented for Mach 1.8 and normalized to sea level.

Table 3.2-1

Reciprocating Engine	0.2# thrust/# engine
Turbojet	8.0# thrust/# engine
Ramjet	30.0# thrust/# engine
Rocket (liquid propellant)	60.0# thrust/# engine

It is obvious from Table 3.2-1 that a reciprocating engine requires a configuration of lift-drag ratio five for support of the engine with other components assumed to be weightless. It will be recalled that five is a reasonably high supersonic lift-drag value. Thus, present reciprocating engines are not adaptable for supersonic flight because of their great weight. Jet engines, on the other hand, develop tremendous thrust per unit weight. It must be recognized that the comparisons possible from Table 3.2-1 would be altered by choice of another flight Mach number. Furthermore, the rocket would appear to even greater advantage if the comparison were to be drawn at great altitude.

Engine Fuel Economy

Given engines which are inherently capable of providing adequate thrust at supersonic speed for low weight, a third factor must be introduced into engine-selection studies. This factor is the fuel-consumption rate of the engine or its inverse, fuel-specific impulse. Intuition is sufficient assurance for the moment that fuel-consumption rate should be low and specific impulse should be high. Figure 3.2-6 presents approximate fuel-specific impulse data for several engine types as a function of Mach number.

It is obvious from Fig. 3.2-6 that wide disparity in specific impulse exists for various engines and Mach numbers. The rocket is least advantageous in this regard; the weight and

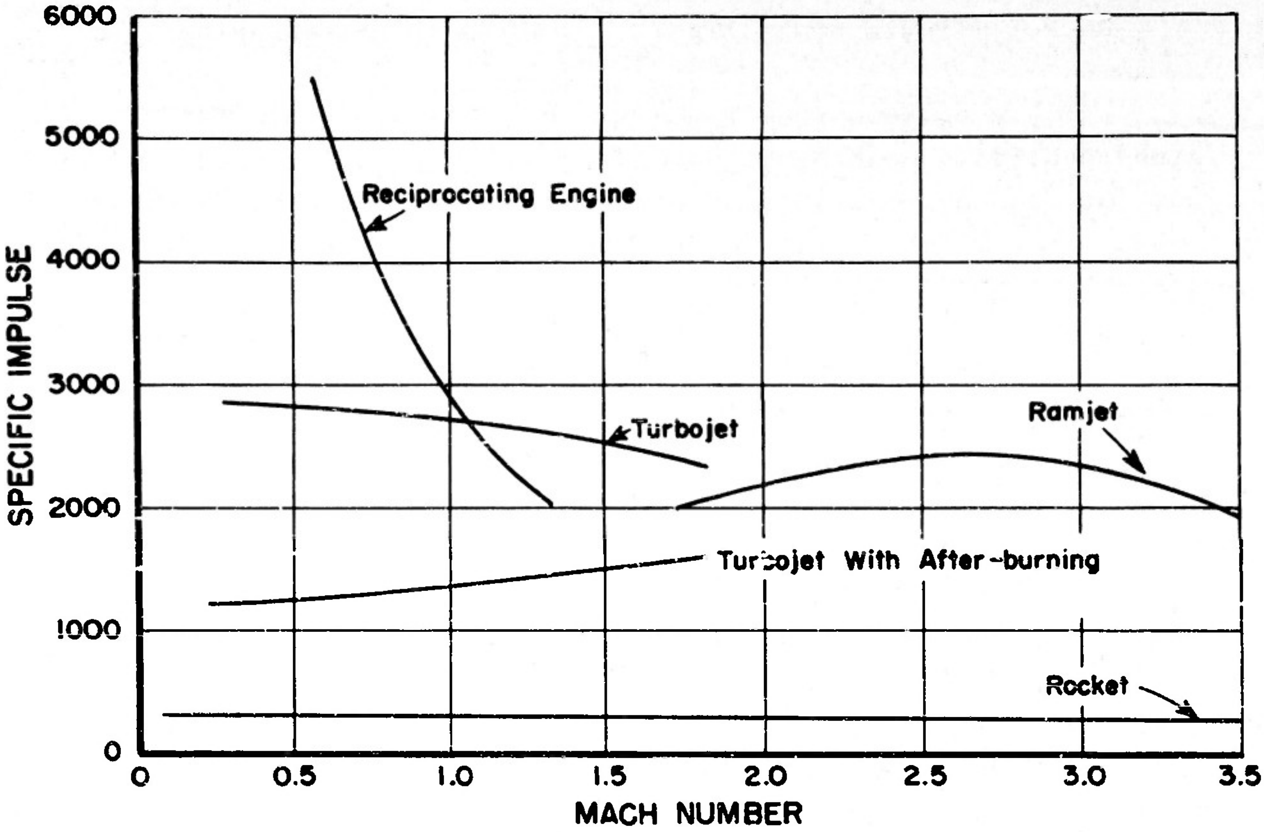

Fig. 3.2-6 TYPICAL SPECIFIC IMPULSE (LB THRUST x SECONDS/LB FUEL) DERIVED FROM VARIOUS TYPES OF ENGINES

thrust advantage evident in Fig. 3.2-5 and Table 3.2-1 for this engine could be negated if a very large quantity of fuel were necessary for range purposes. This would be particularly true if trajectory constraints were such as to prevent optimum operation of the rocket. The ramjet's superior position in the relatively high Mach number range is clearly evident. The reciprocating engine, on the other hand, demonstrates unquestioned superiority at low subsonic speeds. The turbojet fulfills a need for a high subsonic and transonic engine with relatively superior fuel-consumption characteristics. Of course it is well known that the turbojet can also accelerate itself to such speeds, whereas the ramjet cannot, although unassisted acceleration is not always desirable in the composite design sense. If it is recalled that relatively high lift-drag ratios may be obtained through use of swept wings in the low supersonic region, then it is also clear that the turbojet's disadvantageous weight per unit thrust characteristic, when compared to the ramjet, may be mitigated if low supersonic velocity is acceptable by conditions of the problem. Considerations such as these must be fully evaluated in engine-comparison work if it is to be valid, and in general a composite-design study is required. The nature of inter-relations will be discussed in further detail in later paragraphs.

Fuels

It has been implicitly assumed to this point that a hydrocarbon fuel such as gasoline or kerosene was to be used in powerplants. The choice is natural, related as it is to adaptability, availability, and cost. It is also true that hydrocarbon fuels demonstrate general performance superiority over all but a very limited group of possible fuels for air-breathing engines [8]. A few potential fuels, notably boron and the higher borohydrides, offer some promise of higher performance. Diborane and pentaborane among liquid fuels are of interest but possess unfortunate stability and cost characteristics. Among solid fuels, boron promises substantial improvement over, for example, kerosene but suffers from low availability, low purity, except at very high cost, and probably burning and metering difficulties. Decaborane and higher solids appear to have more satisfactory burning qualities than the latter, but present fuel-support problems if solid charges are used. Precautions also need to be taken to prevent spontaneous ignition. Performance in ramjet applications is intermediate between kerosene and boron. Boron carbide appears to be a high-performance fuel, however, considering the present state of the art, combustion of this material will probably be extremely difficult, if not impossible, in a practical engine. Beryllium, a potentially high-performance fuel, has a very poisonous oxide.

Aluminum and magnesium, readily available solid fuels for ramjets, do not compete with kerosene in fuel-economy characteristics. The use of such fuels will in general be limited to short-range application, and in such cases problematical superiority in a given application will depend on the extent to which the weight associated with combustor elements, etc., can be reduced below that required for hydrocarbons. In this field too, the rocket, solid or liquid, will often be competitive. The possibility that novel applications may be discovered, which are dependent on the solid state of these fuels, cannot be discounted, however. They are attractive from a mechanical viewpoint, for example, in the design of an integrated single-stage ramjet and rocket-propulsion system.

The argument for high-fuel density is perhaps the most important reason that fuels other than kerosene are of interest (see Table 3.2-2 and discussion). If fuel-specific impulse is comparable in a given case to kerosene, then obviously less volume need be devoted to fuel with beneficial effects on lift-drag ratio. In general too, it would be expected that tank weight (or fuel surrounding structural weight) would be reduced. If air-specific impulse, likewise, is higher for the same fuel-specific impulse, a reduction in the volume devoted to the engine would also contribute to increased lift-drag ratio and reduced structural weight. Fortunately, for analysis purposes, few fuels are sufficiently superior to kerosene to warrant detailed composite-design studies such as are required to evaluate the above effects.

The effort which has gone into combustion of ramjet fuels other than the hydrocarbons is very limited by comparison. It must be expected, therefore, that progress will be made in this field and that new developments, particularly in adapting the better fuels noted above, may require further evaluation of the position of hydrocarbons in the jet-engine field. Such studies must necessarily include evaluation of the effects of burning on operation of other vehicle components. For example, attenuation of guidance signals may be serious for some fuels and missile-guidance systems. A more detailed survey may be found elsewhere in the text (Chapter 5).

Structural Weight

The structural weight of wings, controls, tanks, etc., occupies a position in analysis similar to that of engine weight; that is, the weight contribution of these parts must be supported by aerodynamic lift. For the supersonic case in which low lift-drag ratio must be accepted, any weight requires an inordinately large amount of thrust compared to the subsonic case with correspondingly large effects on engine weight, fuel load, and tankage requirements. Evaluation of these effects in detail requires lengthy structural computations beyond the scope of the present text. The gross effects will be discussed in later paragraphs. Structural computations are based on well-known structural-design principles to be found in many text-books and other sources [9,10,11]. Special attention is required, however, to the effects of aerodymamic heating on structural properties of materials and thermal stresses on structures, and this subject has been far from exhausted. Pertinent information may be found in various sources [12,13,14].

Staging

Staging is a technique that is used to minimize weight and bulk for a given application. It is called on whenever the range specification requires that a very large fraction of total weight of a given vehicle be devoted to fuel, tanks, and engine. Under this circumstance, it may be advantageous to divide the original vehicle in two or more parts; succeeding parts (stages) thus become the payload of preceding stages. By this device, the necessity for propelling excess weight and waste volume (as fuel is burned) over a long distance is avoided and marked saving under proper conditions may be made. In fact, if the range requirement is sufficiently great, it may be impossible to meet specifications by any alternative.

In the case of the ramjet, staging is a necessity because of the ramjet's inability to deliver static thrust for initial acceleration. The rocket is a natural means for providing necessary initial velocity and functions in this event as the first stage of a two-stage vehicle. (Rockets integral with ramjet structure have been proposed to create a single-stage missile and offer advantages in some applications.)

Additional stages of either rocket or ramjet power are of course possible if the range requirement demands. Simpler forms of staging are also possible; for example, addition of jettisonable wing tanks or belly tanks to an existing configuration provides a means of increasing range, providing the propulsion system can adapt to the initial conditions of operation.

The Range Problem

In the final analysis, judgment as to the worth of any engine and aerodynamic configuration reduces to a problem to determine how large the engine and airframe must be to transport a given payload a given range subject to constraints imposed by guidance, etc. Refinements on this process may be added to arrive at a criterion of relative worth; for example, examination of costs may be required, but the basic problem remains one of composite design.

The first and most obvious factor which must be examined in the design process is related to conditions along the trajectory which the vehicle must follow. For example, air-breathing engines such as the turbojet and ramjet will be limited in altitude by oxygen requirements and minimum combustion chamber pressure (see Fig. 3.2-12) as related to intake conditions from flight through the atmosphere. On the other hand, the rocket may operate satisfactorily in given applications without limitation as to altitude since the oxygen supply is carried aboard. In comparing rockets and ramjets, for example, this factor may be of considerable importance since the rocket, if constrained to a ramjet trajectory, may be penalized. It is necessary, therefore, to consider other trajectory constraints imposed by guidance, permissible velocity variation, aerodynamic heating, terminal flight conditions, etc., in their entirety in order to insure that a fair comparison may be established. Subject to the latter, each powerplant should be operated in the manner most favorable to it.

For through-the-atmosphere applications, the trajectory problem may frequently be reduced to three parts and each part considered separately in the first approximation. The first phase includes launching or take-off and acceleration; the second, cruise flight; and the third, terminal flight. For certain cases (e.g., the Bazooka) the cruise phase may be essentially nonexistent, the trajectory consisting of acceleration and coast phases. For such vehicles, maximum lift-drag ratio may be an academic topic and zero-lift drag of much greater importance. On the other hand, for very long-range, air-breathing engine missiles, the cruise phase not only exists but is by far the most important of the three phases.

The cruise phase of flight is adaptable to analytical investigation and will be considered in some detail as considerable insight into the performance problem may be gained by this means.

It is possible to derive an expression for the flight of atmospheric missiles which includes the basic parameters previously defined and discussed. The expression, first derived by Breguet and named for him, is valid for nonaccelerated flight (although small accelerations do not significantly alter the picture). The differential form, although not subject to acceleration limitations, is particularized in the integration to contain this restriction by virtue of assumption of constant lift-drag ratio, velocity, and fuel-consumption rate.

The derivation is simple and based on the relation that,

$$\frac{dR}{dt} = U = Ma,$$

where

$$dt = \frac{dW}{dW/dt},$$

also

$$\frac{dW}{dt} = \frac{F}{I_f},$$

and as previously noted for equilibrium flight, $\frac{W}{F} = \frac{L}{D}$.

Substitution of these expressions into the derivative of range with respect to time and integration results in,

$$R = I_f \, Ma \, \frac{L}{D} \log \frac{W_O}{W_E}, \qquad (3.2\text{-}3)$$

where W_O and W_E are the initial and final weight conditions, respectively.

The importance of the basic elements of the range problem now becomes clear. Subject to the restraints imposed, the product I_f Ma L/D must be maximized and W_E minimized for maximum range performance. From the previous discussion it is obvious that a simple solution for the problem does not exist, due to the relationship of variables.

It is possible to adapt the range derivative with respect to time to other situations. For example, in cases where lift requirements have a negligible effect on drag, a trivial integration will demonstrate the interdependence of variables.

Elements of Ramjet Missile Performance

Maximization of the range product is subject to various constraints. Perhaps the most important of these is related to provision of an adequate margin of thrust to insure continuance of ramjet flight under various adverse circumstances. Among the factors which make it impossible to design for peak performance must be listed production-engine performance as contrasted to laboratory-controlled performance, adequacy of velocity or Mach number control mechanisms, variation of atmospheric conditions, boost velocity tolerance, and many others. Certain of these are aggravated by specification of high-recovery diffusers which are essential to fuel economy. Such diffusers result in particularly poor ramjet performance under detached shock conditions. The deterioration of performance is associated with reduction of mass flow and pressure recovery and increase in streamline drag of the engine. An enlarged discussion may be found in the chapter on diffusers (Chapter 4).

Figure 3.2-7 has been constructed to demonstrate the effect of diffuser (Ferri) design point, air-fuel ratio, and Mach number on ramjet thrust coefficient. For reference purposes, the drag coefficients associated with cruise and zero lift for a typical configuration have been superimposed on the thrust coefficient plot. The cruising drag coefficient has been matched to the thrust coefficient at $M = 2.0$ for diffuser on design, specified nozzle geometry, and engine operation at an air-fuel ratio of 45. At higher or lower Mach number, richer mixtures must be used to provide cruising thrust requirements. Obviously, fuel-consumption rate will increase under these conditions. It is particularly important to observe that doubling the fuel flow does not double thrust coefficient at $M = 2.0$; in fact, the increase is only about 35 per cent. Little additional thrust can be obtained by burning at stoichiometric air-fuel ratio. (Constant burning efficiency over the above range of air-fuel ratios has been assumed for purposes of discussion.) This condition results from the fact that the diffuser shock configuration for an air-fuel ratio of 23 (or 15) results in reduced mass flow at the entrance to the engine and increased drag in the region of the cowling. The missile designer must, therefore, determine if the flight conditions which the vehicle will encounter require a margin

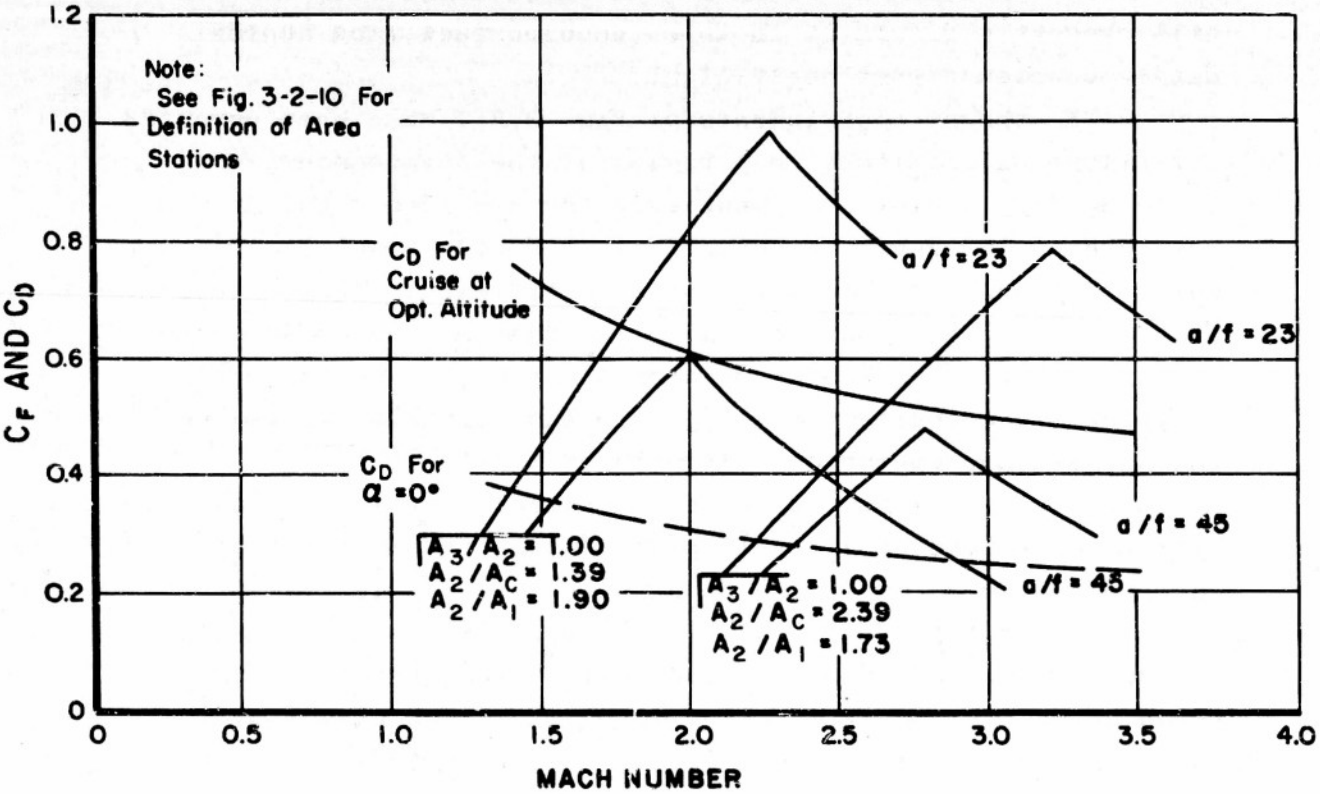

Fig. 3.2-7 THRUST COEFFICIENT (C_F) AS A FUNCTION OF MACH NUMBER, RAMJET GEOMETRY, AND AIR-FUEL RATIO. DRAG COEFFICIENT (C_D) OVERLAY FOR CRUISING AND ZERO LIFT FLIGHT CONDITIONS FOR A TYPICAL CONFIGURATION

(n_b = 0.8)

of thrust of greater than 35 per cent. If a greater value is required, the most effective way to achieve it is to operate the diffuser off design in the swallowed shock regime for cruise conditions. Referring again to Fig. 3.2-7, it is evident that operation of the previous ramjet engine at $M = 2.3$ will present a situation in which doubled fuel flow approximately doubles thrust coefficient.

The thrust coefficients of Fig. 3.2-7 have been computed for flight in the isothermal region of the stratosphere ($t = -67^{o}F$). A similar computation for sea-level ambient temperature would result in appreciably-reduced thrust coefficients. Fortunately, however, thrust coefficient requirements are usually appreciably less also, so that engine performance (for on design or slightly swallowed shock diffuser operation) is generally adequate for a high-altitude missile constrained to low-altitude operation. Effects of missile weight component, as in climbing flight, are also minimized under low-altitude flight conditions as a result of the large absolute thrust which is obtained from the engine under high air density conditions.

Comparison of Fig. 3.2-7 and Fig. 3.2-1 will reveal that the ramjet engine cannot provide equilibrium thrust for the assumed aerodynamic configuration for an angle of attack of fourteen degrees. The designer must insure, therefore, that demands on the missile do not cause an excessive angle of attack which is beyond the capability of the engine to counteract. It may be possible, however, to tolerate this condition for a short period of time in a particular situation, e.g., near the end of flight.

Burning efficiency is generally not constant with air-fuel ratio, as assumed in preceding paragraphs. It may be peaked for lean mixtures or for rich mixtures, depending on the design criterion for a particular missile. For example, a long-range missile will generally require an engine developing peak efficiency at lean mixtures; an interceptor-type missile, on the other hand, may require peak efficiency at high thrust coefficient for climb and maneuverability reasons. Burning efficiency will also vary with Mach number and altitude because of engine inlet stagnation temperature and pressure variations. It will vary, particularly in the swallowed shock regime, because of changing Mach number at the entrance to the burner (associated drag effect in addition). The interpretation with respect to Fig. 3.2-7 of these various factors is simply to change the apparent fuel-air ratio required to produce the desired thrust coefficient. A limitation exists, however, on the extent to which this statement is applicable; i.e., it is not possible to increase thrust coefficient beyond the value associated with maximum air-specific impulse for a given engine. This is discussed in greater detail elsewhere in the text (Chapter 4).

Mach number tolerance is not great at less than design Mach number for ramjet engines. For the previous example, insufficient thrust is developed by the engine at a Mach number 20 per cent less than design Mach number to cruise the vehicle. Should the missile decelerate through a combination of circumstances to this point, continued flight would only be possible if the missile attitude were changed to reduce the drag coefficient and increase the missile weight component contributing to acceleration to regain near-design point operation. The situation becomes more critical as design Mach number is increased, as may be seen in Fig. 3.2-7.

Figure 3.2-8 has been constructed to demonstrate the variation of fuel-specific impulse with Mach number and air-fuel ratio for the ramjet geometries of Fig. 3.2-7. Operation off-design, either from the air-fuel ratio standpoint or the Mach number viewpoint, is disadvantageous to fuel-consumption characteristics of the ramjet engine. For short-range missiles (i.e., for those missiles whose fuel load is a small fraction of weight) this factor may not be overly important. For long-

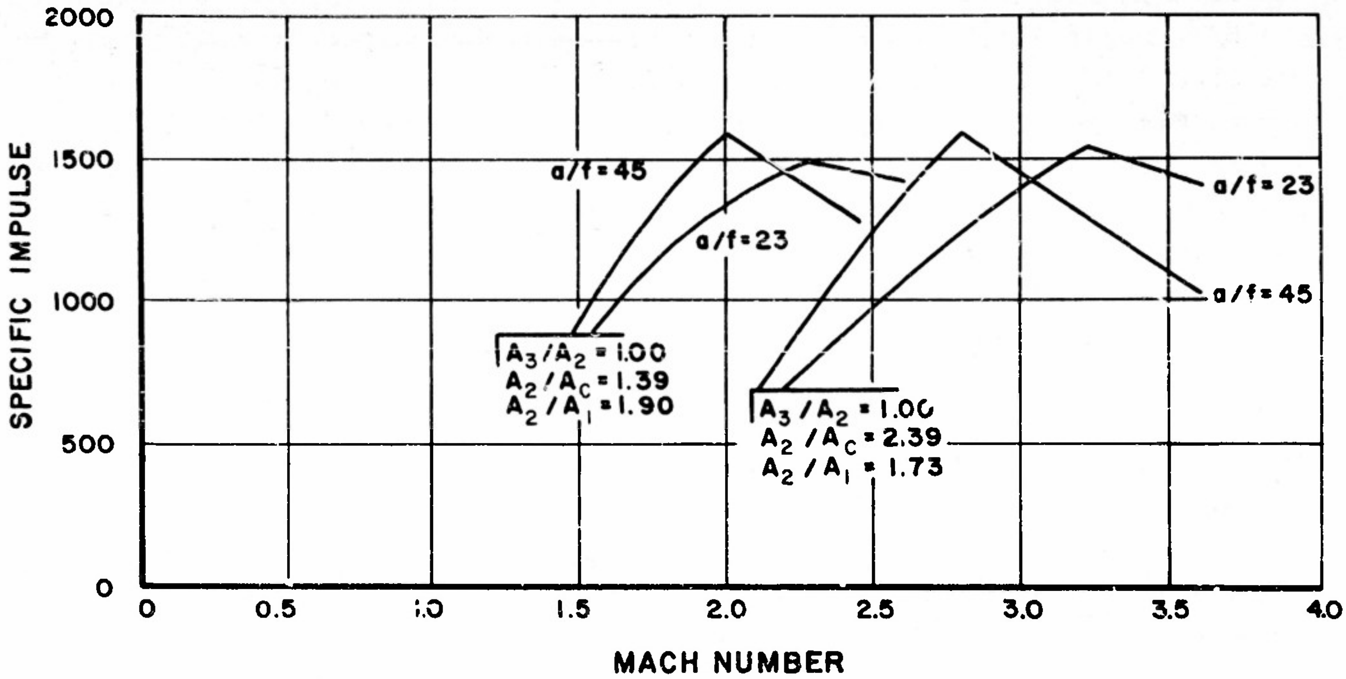

Fig. 3.2-8 SPECIFIC IMPULSE FOR THE RAMJET GEOMETRIES OF FIG. 3.2-7 AS A FUNCTION OF MACH NUMBER AND AIR-FUEL RATIO

(n_b = 0.8)

range missiles, the loss in fuel may be serious and every effort should be exerted to minimize the loss.

Long-range missiles typically operate at very lean air-fuel ratio. It is possible, therefore, to conduct a design point analysis (i.e., diffuser on design) for various combinations of ramjet variables with assurance that a fair margin of thrust over drag will result for rich mixtures. (An exception exists for diffusers which experience rapid deterioration of pressure recovery in the subcritical region.) Such an analysis can be very instructive in determining desirable characteristics for the propulsion and airframe system.

Figure 3.2-9 presents results of a design-point analysis for M = 2.5. The independent variable is equivalence ratio, i.e., the ratio of the specified fuel-air ratio to the stoichiometric fuel-air ratio. (Stratospheric heights have been assumed for this and succeeding curves of similar nature.) It is obvious that optimization of specific impulse requires lean burning and expansion of the jet to ambient pressure. It is also apparent that reduction of drag, therefore thrust requirement, is beneficial not only from an external aerodynamics point of view (i.e., in increasing lift-drag ratio) but also from an engine optimization point of view. It follows also that the combustion-chamber diameter should be designed to the maximum value permitted by the surrounding structure to achieve maximum specific impulse. Such a procedure is favorable as well to provision for an adequate margin of thrust and to minimization of tailpipe temperatures. Reduction of the latter can be particularly significant not only from a structural standpoint but also from a mixing and therefore burning efficiency point of view. Maximization of the burner diameter, however, is contrary to minimization of external drag, subsonic diffuser length and weight, etc. Again compromise is indicated as necessary.

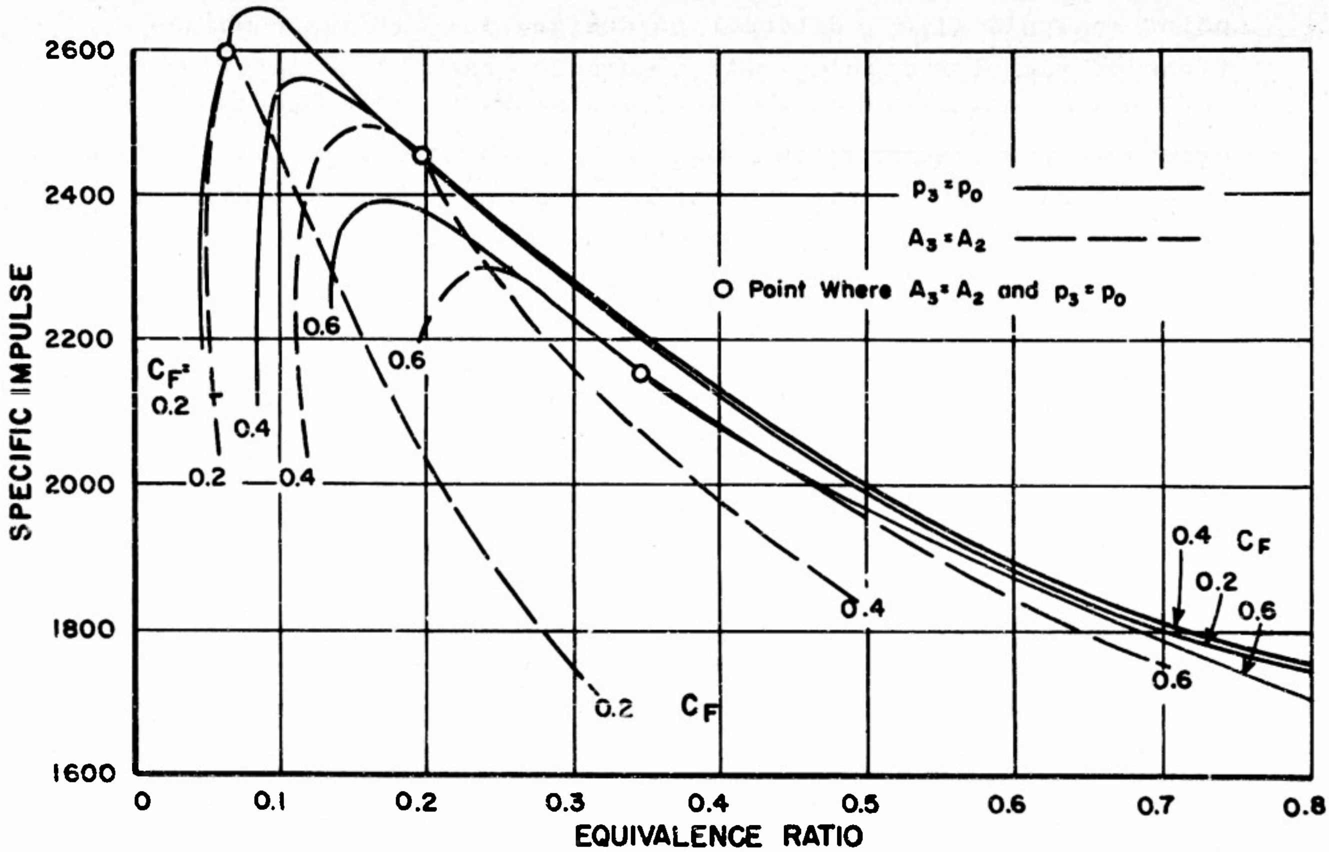

Fig. 3.2-9 SPECIFIC IMPULSE AS A FUNCTION OF EQUIVALENCE RATIO, THRUST COEFFICIENT, AND NOZZLE CONFIGURATION. DESIGN POINT ANALYSIS

$C_{Db} = 1$; $n_b = 100\%$; $T_o = -67^{o}F$; $M_o = 2.5$

Figures 3.2-10 and 3.2-11 present the ramjet geometry corresponding to Fig. 3.2-9. It is apparent from Fig. 3.2-10 that ramjet geometry can have a decided effect on external drag. For example, rich burning requires that the diffuser entry be small with respect to the combustion chamber. Thus drag associated with spillover and the immediate lip of the diffuser will be relatively less than for larger intakes. The transition section from the intake to the combustion chamber will cause relatively greater drag, however, and must be balanced against the reduction of the intake drag. Packaging considerations, obviously, will play an important part also. Detailed studies are required in a particular instance to insure optimum operation of the whole. A similar situation exists at the exit. In this instance, a balance is required between the drag associated with cone, base, or boattail and the nozzle geometry which results in maximum specific impulse.

Figure 3.2-12 indicates the magnitude of diffuser exit Mach number corresponding to the geometry of Fig. 3.2-11. It is perhaps obvious that this value should be low to minimize burner length, burner drag, etc.

In studies of the performance of the ramjet engine as a function of the independent variable Mach number, it is frequently advantageous when maximizing range to consider the effect of various parameters on the product of specific impulse and Mach number, rather than the effect on specific impulse alone. This follows from the Breguet equation and the fact that lift-drag ratio for ramjets generally is not strongly dependent on Mach number whereas specific impulse frequently is quite dependent, thus optima exhibited by the product tend to be more significant than specific impulse optima.

The effect of various diffuser pressure recovery values on the product of specific impulse and Mach number for a typical thrust coefficient requirement is shown in Fig. 3.2-13.

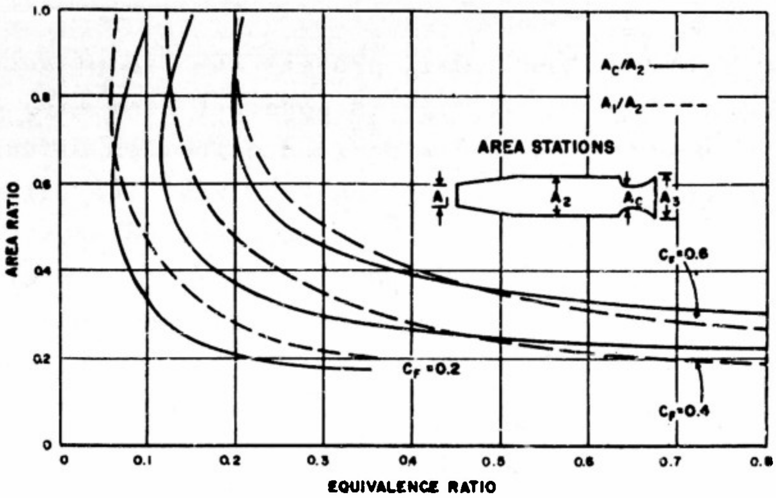

Fig. 3.2-10 RAMJET GEOMETRY AS A FUNCTION OF EQUIVALENCE RATIO AND THRUST COEFFICIENT. EXIT AREA (A_3) EQUAL BURNER AREA (A_2)
(SEE FIG. 3.2-9 FOR CORRESPONDING SPECIFIC IMPULSE)

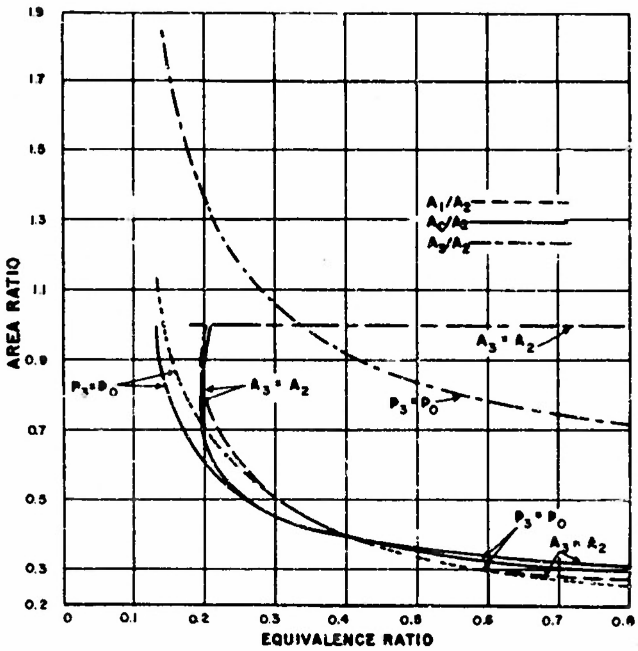

Fig. 3.2-11 RAMJET GEOMETRY AS A FUNCTION OF EQUIVALENCE RATIO AND NOZZLE CONFIGURATION FOR $C_F = 0.6$.
(SEE FIG. 3.2-9 FOR CORRESPONDING SPECIFIC IMPULSE)

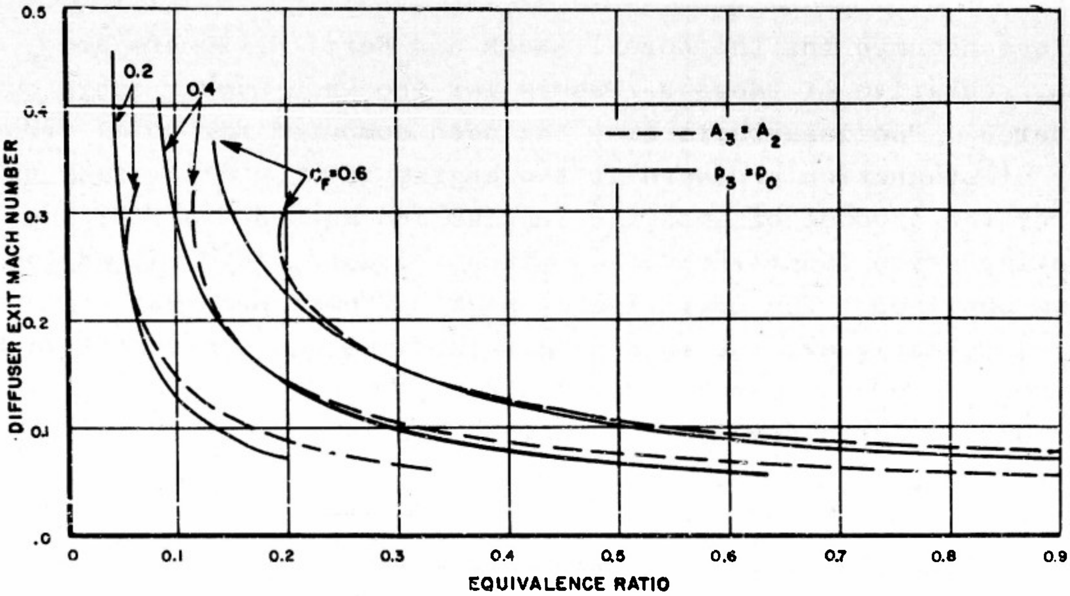

Fig. 3.2-12 DIFFUSER EXIT MACH NUMBER AS A FUNCTION OF EQUIVALENCE RATIO AND NOZZLE CONFIGURATION FOR C_F = 0.2, 0.4, & 0.6.

(FIG. 3.2-9 FOR CORRESPONDING SPECIFIC IMPULSE)

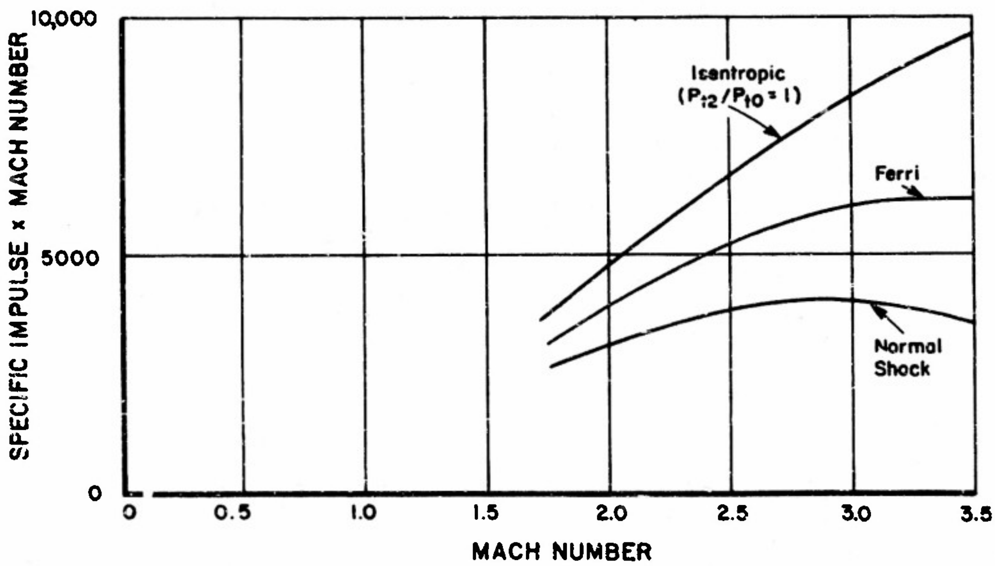

Fig. 3.2-13 SPECIFIC IMPULSE - MACH NUMBER PRODUCT FOR A TYPICAL RAMJET AS A FUNCTION OF MACH NUMBER AND DIFFUSER TYPE (RECOVERY)

C_{Db} = 1; n_b = 0.8; $p_3 = p_o$; T_o = -67°F

Values assumed for the normal shock and Ferri diffusers are representative of these diffusers for the Mach number range considered. The isentropic case has been computed for total recovery of stagnation pressure at the engine inlet. A limiting value of the product of specific impulse and Mach number for the configuration (and listed assumptions) under consideration is thus obtained. The advantage of high diffuser pressure recovery is clearly evident in Fig. 3.2-13. The association of Mach number and pressure recovery should be noted. Definite maxima for particular diffusers are shown. The importance of this factor must be considered in terms of maximization of the product I_f M L/D. More subtle, but nevertheless important effects associated with diffuser pressure recovery include effects on external geometry, therefore drag, and on temperature in the combustion chamber. The latter may be particularly important if thrust coefficient demands are high.

The effect of diffuser drag has not been considered in the present text because of the complex relationship which exists between this factor and external aerodynamics (optimum angle of attack, flight altitude, etc.). The importance of this factor is related not only to external contours of the diffuser, but to the mass flow acceptance ability of the diffuser when operated on design. It is characteristic of many high-recovery diffusers that full air flow based on frontal area is not accepted even when operation is at design conditions. Thus comparison of such diffusers with other types which accept full mass flow is necessarily intricate. The influence of engine operation on this characteristic also must be investigated and properly assessed. It is generally convenient in this regard to separate the spillover drag into two components, one of which should be considered a tare value and added to external aerodynamic drag. This tare is associated with diffuser operation at maximum mass flow acceptance. A second component, that associated with mass flow acceptance of less than maximum, is related to engine operation and should be treated as a decrement on engine thrust.

A second factor of great significance in ramjet performance is related to nozzle design. Figure 3.2-14 demonstrates the extent to which the specific impulse, Mach number product for a chosen diffuser and missile can be influenced by nozzle geometry and Mach number. The case $A_2 = A_3 = A_c$ represents conditions for a straight tailpipe. Performance deteriorates as flight Mach number is increased and the disparity between flight and jet exit Mach number increases. It is also characteristic of the straight tailpipe that A_1 exceeds A_2 at high Mach number. An additional interaction with external aerodynamics takes place by virtue of this fact, and the result is an increase in boat-tail and/or base drag as flight Mach number is increased. It is thus evident as a minimum measure that A_3 should be increased relative to A_1 and A_2. The effect of this action on drag is not shown; however, the effect on the specific impulse, Mach number product is demonstrated by the case $A_1 = A_3 \geqslant A_2 = A_c$.

A further increase in the specific impulse, Mach number product can be obtained by addition of a throat, as indicated by the plot of this variable for the condition that jet exit pressure equals ambient pressure ($p_3/p_0 = 1$) and $A_3 = A_2$. Reference [15] demonstrates that the condition $p_3/p_0 = 1$ will afford maximum specific impulse for a given flight Mach number. The dimensions of the required constrictor and jet exit Mach number will, of course, be influenced by diffuser pressure recovery and duct losses.

Figure 3.2-15 introduces two additional factors, burner drag and burner efficiency, which influence specific impulse. It is, of course, obvious that drag should be low and efficiency high. It is of interest to note that if the number of combustion chamber entrance velocity heads were reduced from five to one, the subsequent change in engine performance would be equal to a

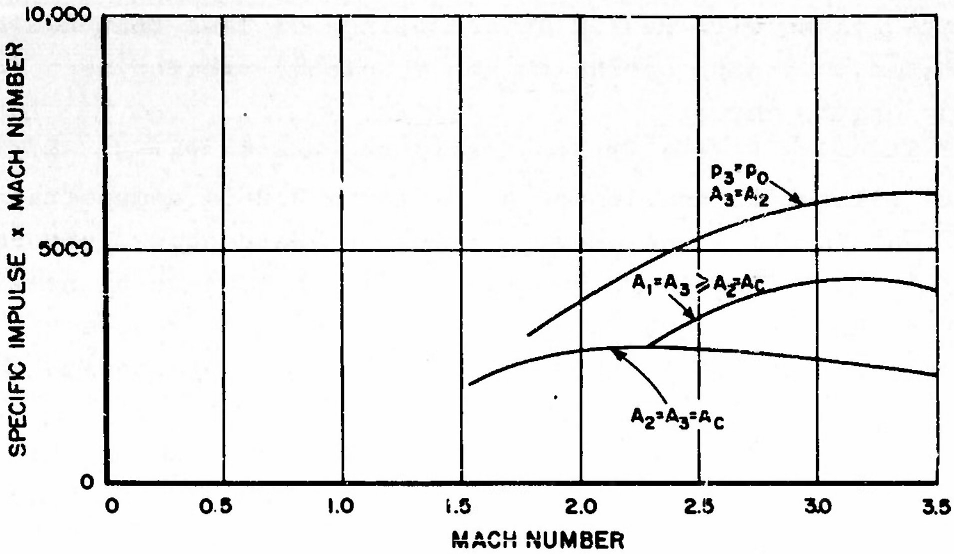

Fig. 3.2-14 SPECIFIC IMPULSE - MACH NUMBER PRODUCT FOR A TYPICAL RAMJET AS A FUNCTION OF MACH NUMBER AND DUCT AREA RELATIONSHIPS

Ferri Diffuser

$C_{Db} = 1;\ n_b = 0.8;\ T_o = -67^{\circ}F$

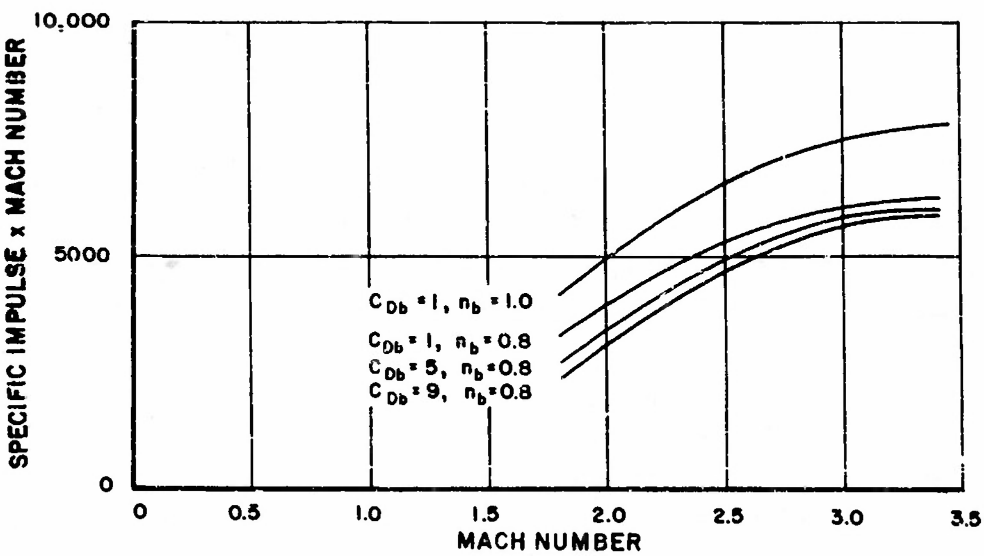

Fig. 3.2-15 SPECIFIC IMPULSE - MACH NUMBER PRODUCE FOR A TYPICAL RAMJET AS A FUNCTION OF MACH NUMBER, BURNER DRAG, AND BURNER EFFICIENCY

Ferri Diffuser

$p_3 = p_o;\ T_o = -67^{\circ}F$

twenty per cent improvement in burner efficiency at Mach 1.8, and that the drag becomes a progressively smaller fraction of equivalent burner efficiency as Mach number is increased. Obviously, therefore, an increase in burner drag to increase burner efficiency is likely to be more acceptable at high Mach number than at low. Burner drag may also affect external drag through its effect on the ratio A_1/A_2. As a result of the associated change in thrust coefficient requirements, secondary effects on specific impulse may be anticipated.

Interactions between the ramjet engine and aerodynamics are frequently of a subtle nature. One example may be found by reference to Figs. 3.2-3 and 3.2-16. The latter illustrates the effect of Mach number and diffuser pressure recovery on the maximum allowable altitude of flight for a limiting combustion chamber pressure. (The limitation is related to engine volume and burning efficiency, and is discussed in Chapters 8, 9, 10, and 11.) If one-half atmosphere pressure is required, then it is evident that the difference in maximum flight altitude for different diffuser recovery efficiencies may amount to many thousands of feet. For example, at $M = 2.5$, the limiting flight altitude for the Ferri diffuser is 71,500 feet, but for isentropic compression the limiting altitude would be increased to 78,000 feet. Turning to Fig. 3.2-3, it will be noted that operating lift-drag ratio improves from a value of 4.85 at the lower altitude to 5.15 at 78,000 feet. Thus in addition to a primary improvement in specific impulse which results from isentropic compression, a secondary advantage accrues from an accompanying increase in permissible flight altitude and associated increase in operating lift-drag ratio. This argument has been based on an increase in diffuser recovery without increase in diffuser drag. Since many high recovery diffusers exhibit increased drag, this factor must be integrated as well in detailed studies. Two effects will be observed as a consequence. Maximum lift-drag ratio will be reduced, but optimum

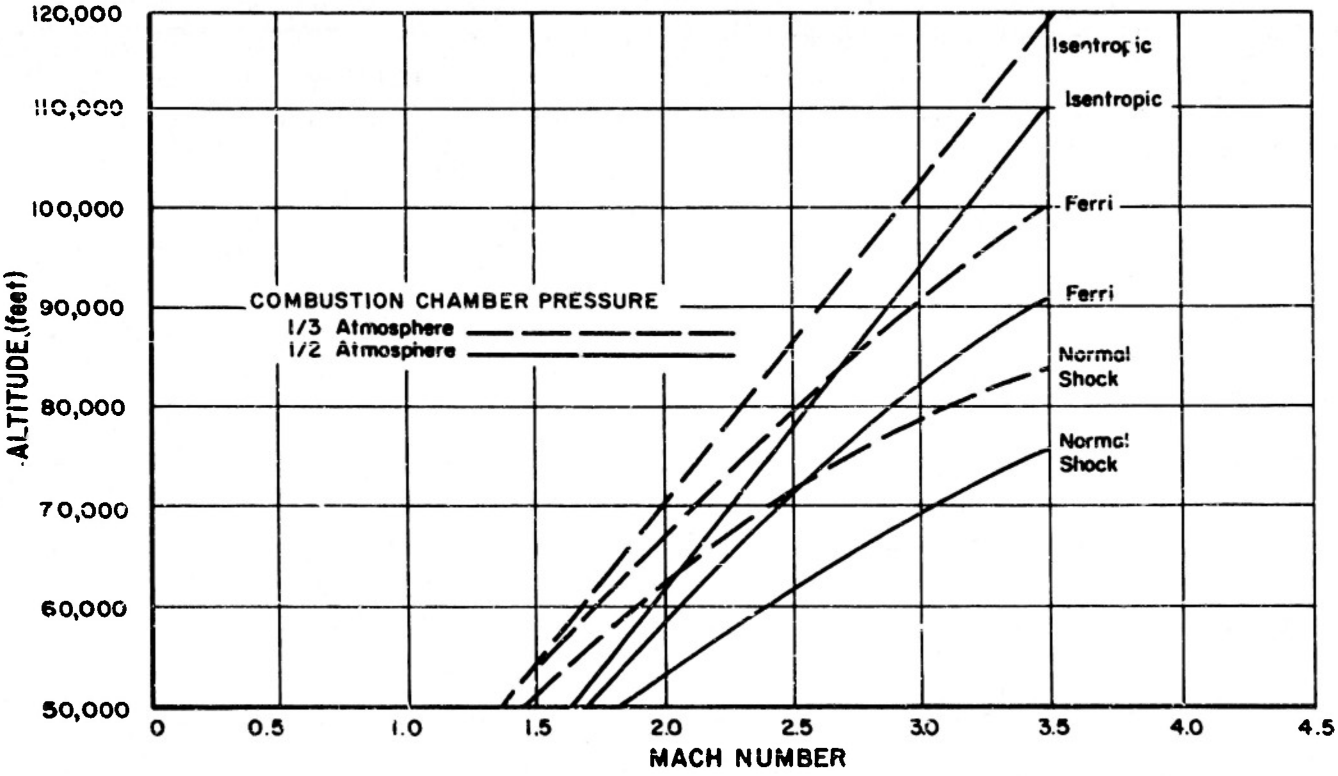

Fig. 3.2-16 MAXIMUM FLIGHT ALTITUDE AS DETERMINED BY DIFFUSER PRESSURE RECOVERY AND CONSTRAINTS ON MINIMUM ACCEPTABLE COMBUSTION CHAMBER PRESSURE

angle of attack will be increased, with a further demand for maximum operational altitude.

It will be recalled that assumptions made in the derivation of the Breguet range expression were tantamount to assumption of a constant lift-drag ratio, therefore, constant angle of attack, throughout flight. Combustion chamber pressure limitations may constrain a missile to an altitude less than the optimum value. For a Breguet trajectory, the end point of cruise flight (i.e., near minimum fuel load) will coincide with the maximum permissible altitude; the beginning of cruise flight may, therefore, take place at much lower altitude. The required wing area under this condition may be substantially less than optimum from the external point of view. In this circumstance, it is desirable to follow other than a Breguet trajectory to permit increase of wing area with a consequent improvement in average lift-drag ratio for the total flight path. All or part of the cruise flight may be visualized to occur at the maximum altitude permitted by the combustion chamber pressure limitation. The missile angle of attack will vary during flight under these conditions with a consequent demand for satisfactory diffuser operation over the indicated angle of attack range. In particular instances, it may be found necessary to insure constancy of diffuser attitude by use of wing-control configurations.

A further factor implied above which enters the picture is the relation of burning efficiency to combustion chamber pressure. Figure 3.2-3 indicates that an altitude of approximately 85,000 feet would produce a maximum lift-drag ratio of 5.5 at $M = 2.5$. Actually, on the basis of Fig. 3.2-4, this probably represents near absolute maximum in lift-drag ratio and associated flight altitude for this Mach number. In Fig. 3.2-16 it may be seen that nearly isentropic pressure recovery is required to produce one-third atmosphere in the combustion chamber at 85,000 feet altitude. If the percentage drop in burning efficiency is greater than the percentage increase in lift-drag ratio at this altitude compared to lower altitudes, then it is obvious that a better compromise can be found at a lower altitude.

Burning efficiency can be maintained at a high level at very low combustion chamber pressures by increasing tailpipe length. The missile designer, however, must balance the gains to be achieved by very high altitude flight against the losses entailed in increasing engine size. It is perhaps worth noting from the included figures that achievement of high burner efficiency at one-third atmosphere combustion chamber pressure and near-isentropic diffuser pressure recovery would permit near maximum possible lift-drag ratios to be utilized for $M = 2.5$ or higher missiles. A similar condition has been observed in other studies as well, so that at least approximately such limits represent useful goals to work toward in developing diffusers and minimum volume burners of maximum usefulness for long-range missiles.

Figure 3.2-17 presents a general solution of the Breguet expression for a particular payload and two range and structural weight assumptions. The solution is applicable to any type of engine. In fact, the concept is valid for a case as unrelated to aircraft as the locomotive and its cars, wherein an enormous "lift-drag" ratio and high "specific impulse" permits large loads to be carried a considerable distance with small fuel expenditure. Certain generalizations may be made on the basis of Fig. 3.2-17. It is clearly evident that great reduction in missile weight is possible for the 1250 nautical mile missile if the product I_f M L/D equals 25,000 rather than 8000-10,000, but that improvement beyond this point results in progressively reduced returns. It is also evident that the value of I_f M L/D which must be achieved to get beyond the

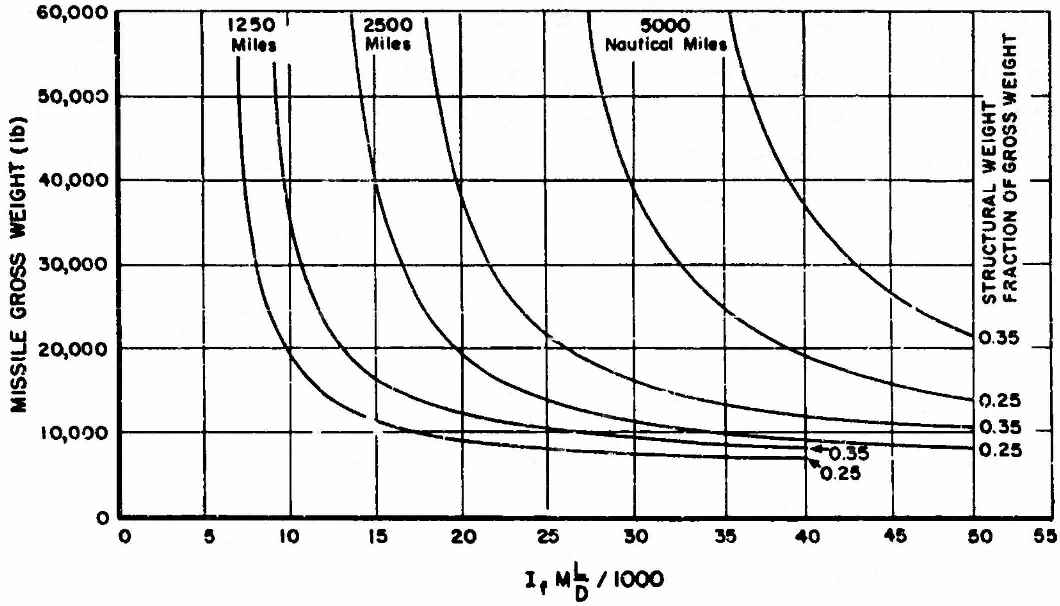

Fig. 3.2-17 MISSILE GROSS WEIGHT AS A FUNCTION OF RANGE, STRUCTURAL WEIGHT FRACTION OF GROSS WEIGHT, AND THE PRODUCT OF LIFT-DRAG RATIO, SPECIFIC IMPULSE, AND MACH NUMBER. PAYLOAD = 4000 POUNDS

"knee" of the curve becomes progressively larger as the range requirement is increased. Similar reasoning attends changes in structural weight fraction of gross missile weight. In fact, for the 1250 nautical mile missile it is probable that reduction of structural weight is a more promising means for reducing the size and weight of the missile than increasing I_f M L/D beyond 25,000. In a given instance compromising the units which comprise the product I_f M L/D may be justified if the proposed changes reduce structural weight sufficiently.

Reference to Figs. 3.2-3, 3.2-4, and 3.2-13 indicates that values of I_f M L/D of 25,000-30,000 are obtainable by high-altitude ramjets flown at a Mach number of 2.5, a Mach number not incompatible with use of aluminum structures at high altitude. For a range objective of 2500 miles, values of this order are sufficient to exhaust the enormous reduction in gross missile weight possibilities existent in increase of I_f M L/D from lower values. This is true for two values of structural weight fraction. The values chosen have not been selected to define a particular missile but to demonstrate the principle involved. They do tend to bracket reasonable values of structural weight fraction for the range of variables considered. In general, it is found that increase of Mach number and reduction of range will increase the magnitude of the structural weight fraction of gross missile weight.

For 5000-mile vehicles substantially higher values of I_f M L/D and the lowest possible structural weight fraction are critically desired to reduce missile weight. Even small improvements can be very significant at this range. Again referring to Figs. 3.2-4 and 3.2-13, it is seen that desired values of I_f M L/D can only be obtained by operation at Mach numbers of three or greater, and that diffusers must be capable of near-isentropic compression at such Mach numbers. Thus one may regard the 1250-mile ramjet missile as easily obtainable on the basis of existing information, and be secure in the knowledge that a nearly optimum solution will be obtained. For the 5000-mile case, the need for superior characteristics in all departments of missile design is much more pressing. The argument, of course, is not meant to rule out obtaining maximum performance at whatever range is required, but simply is meant to place the matter in the proper perspective. Thus, for example, great effort would not be warranted to improve missile performance for the 1250-mile case, whereas such effort might be imperative for the 5000-mile example. (The maximum value of I_f M L/D shown in Fig. 3.2-17 is based on turbulent boundary layer lift-drag ratios and Fig. 3.2-13 specific-impulse values. If laminar flow can be maintained over large areas, as previously discussed, considerable benefit will accrue to both 1250- and 5000-mile missiles but particularly to the latter.)

It is clear from Fig. 3.2-17 and the preceding discussion that attainment of low gross weight is bound closely with the attainment of low structural weight and high I_f M L/D; furthermore, that demands on these parameters grow more insistent as range is increased. An alternative to improvement of these items may be found in the staging principle. Figure 3.2-18 has been constructed to compare two stages of ramjet power with one stage. For expected values of I_f M L/D etc., a second ramjet stage is disadvantageous if the range objective is 2500 nautical miles. However, for 5000 miles the decision is not as clear cut. If design considerations prevent attainment of an I_f M L/D of 30,000 or more and low structural weight fraction, staging must be regarded as a distinct possibility for achieving minimum gross weight. For greater ranges the advantages of staging would become even more apparent.

Reference has been made in previous paragraphs to the part that increased fuel density can play in increasing missile range, or in minimizing weight for a given range. Table 3.2-2

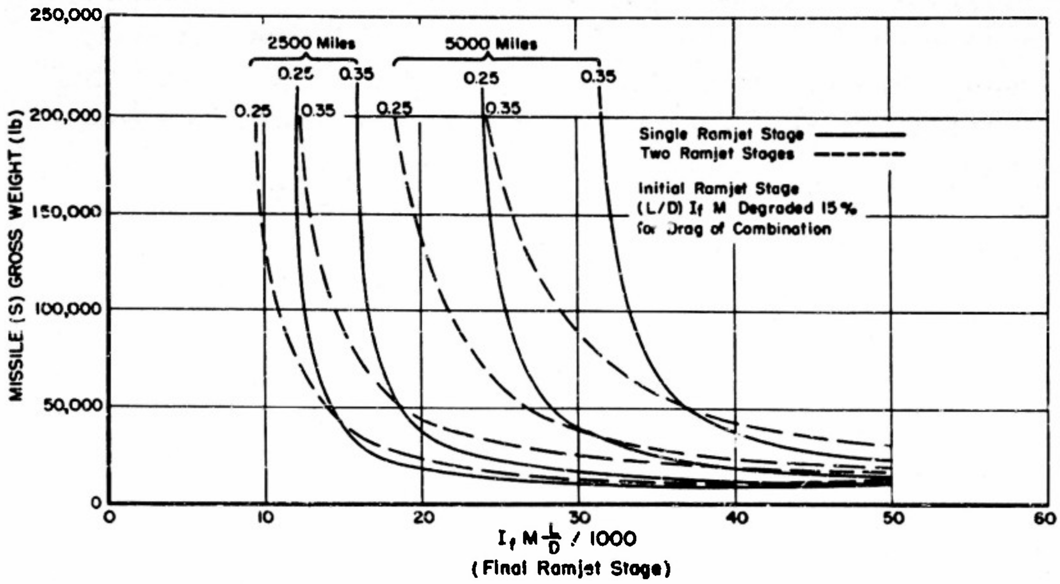

Fig. 3.2-18 MISSILE(S) GROSS WEIGHT AS A FUNCTION OF RANGE, STRUCTURAL WEIGHT FRACTION OF STAGE WEIGHT, PRODUCT OF LIFT-DRAG RATIO, SPECIFIC IMPULSE, AND MACH NUMBER, AND NUMBER OF STAGES OF RAMJET POWER

demonstrates the correspondence which exists among fuel density, fuel specific impulse, and range for a long-range missile of fixed tankage.

Table 3.2-2

Effect of Fuel Density and Specific Impulse on Range
(Effect of variables on structural weight neglected)

Normalized Density	Normalized Specific Impulse	Range
1.0	1.0	2500
1.2	1.0	2840
1.0	1.2	3000
1.2	1.2	3410
1.2	0.8	2280

It is evident that a 20 per cent increase in specific impulse is to be preferred to a 20 per cent increase in density. In actual fact the differences given in Table 3.2-2 would be somewhat more unfavorable to the increased density examples if structural weight were increased to maintain a specified design load factor. Other effects on structure caused by substitution of different fuels cannot be evaluated except by detailed analysis beyond the scope of this chapter. It should be noted that normalization of the above examples to the base range (2500 nautical miles) would be subject to the logarithmic behavior of the range formula which was previously discussed. It would be expected, therefore, that the resulting differential existing between the above examples would be even greater than apparent from a consideration of the various ranges. In such a comparison, the relative advantage of increased fuel specific impulse as contrasted to increased density would be magnified.

Ramjet Acceleration Phase

The minimum two-stage characteristic of the ramjet missile requires concurrent study of the rocket stage. Figure 3.2-19 presents data on typical rocket (solid própellant) requirements to achieve a specified Mach number. From the magnitude of the quantities involved it may be concluded that the rocket stage is a large part of the total vehicle. This factor has obvious significance from handling, etc., viewpoints.

It is necessary to consider previous [Fig. 3.2-17] arguments when choice of boost Mach number is required in a specific instance. For example, for 1250-mile missiles it was noted that I_f M L/D value of greater than 25,000 produced progressively less advantage. Thus it might be expected that the advantage to be gained by Mach number in excess of that required for I_f M L/D = 25,000 would be outweighed by the increase in rocket size required to produce cruise Mach number at the end of boost. For the 5000-mile single ramjet stage case, the situation is different. From previous arguments it is clear that higher values of I_f M L/D are greatly to be desired in the interests of reducing missile weight, and that required values may be obtained by very high Mach number flight and near-isentropic compression diffusers. If adequate diffusers can be developed it is found that reduced missile weight at increased Mach number more than offsets increased rocket weight to produce the Mach number. Thus it is clear that rocket boosters represent a larger fraction of total launched weight as missile range increases, and that this is related to an increase in optimum flight Mach number as design range is increased.

The above argument is related primarily to the use of efficient high Mach number diffusers. However, it is also true for missile systems operating near their absolute maximum range capability, that increase of booster impulse and boost

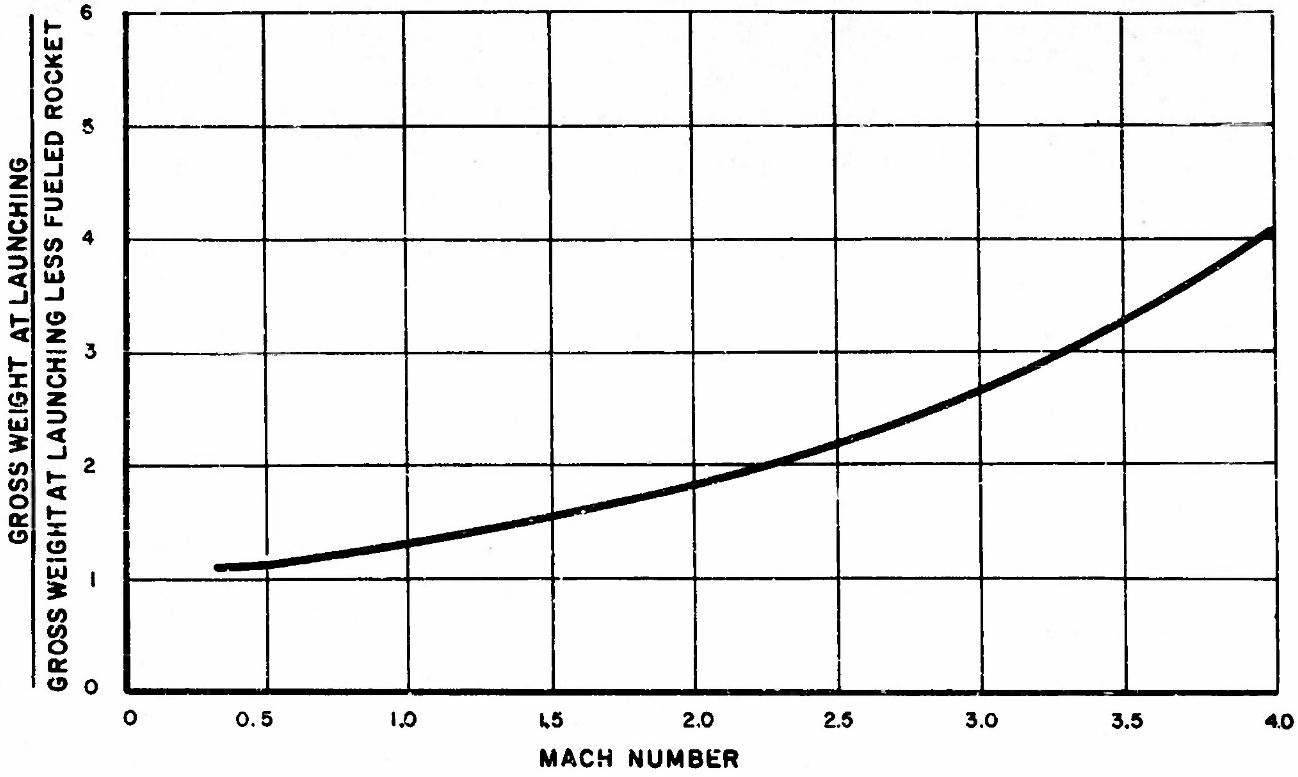

Fig. 3.2-19 THE RATIO OF GROSS WEIGHT AT LAUNCHING TO WEIGHT OF MISSILE, BOOSTER, FINS, AND ATTACHMENTS AS A FUNCTION OF BOOST MACH NUMBER REQUIRED FOR A TYPICAL RAMJET PLUS BOOSTER CONFIGURATION (SINGLE STAGE, HIGH ACCELERATION BOOST)

Mach number can be optimum from the staging point of view without regard for the diffuser. This condition would exist, for example, in the event that a relatively high launching acceleration is required and rocket thrust is not variable. It is possible, of course, that a coast period after boost might be desirable in this case.

The use of variable exits, bleed, or inlets for ramjet engines can be effective in reducing booster requirements to achieve a specified Mach number. The desirability of such a measure is related to the staging problem. If the range requirement is very great and the initial missile fuel fraction of weight very large, then it may not be desirable to use such devices but rather to contain the necessary impulse in the jettisoned first stage. If the fuel fraction is small, the weight (in missile and booster) and complication of the variable devices must be weighed against fuel savings which they permit.

Additional stages of rocket boost can be effective in reducing launching weight under proper conditions. The situation in this regard is not unlike that found previously in derivation of the Breguet range equation, i.e., a logarithm is found in the equation for boost velocity. Consider the simple drag-free expression for terminal velocity.

$$F = \frac{W}{g}\dot{U} = \frac{\left(W_o - \frac{F}{I_f}t\right)}{g}\dot{U} \qquad (3.2\text{-}4)$$

Integration of the expression and substitution of W_O and W_E for initial and final weights respectively, results in:

$$U = g\, I_f \log \frac{W_O}{W_E} \qquad (3.2\text{-}5)$$

The similarity of this expression and the Breguet expression in a mathematical sense permits the conclusion that factors which affect staging will be similar in the two cases. Terminal velocity, fuel specific impulse, and booster metal parts weight will define conditions wherein staging may be advantageous. In the actual case, drag will introduce additional factors such as fuel density and aerodynamic configuration characteristics. The effects of launching angle will be several. The missile and booster weight component will have a subtractive effect on terminal velocity with magnitude related to acceleration, however, the resulting flight path will be such as to reduce air density (i.e., drag) and the speed of sound with salutary effects on terminal Mach number.

The acceleration phase of ramjet flight may be treated for short duration booster rockets by simple analytical means by assuming drag to be a function of time, an assumption which is very nearly true in many practical cases. Long boost time or acceleration and climb on ramjet power, however, introduce complications into the analysis which defy analytical procedures, except at great cost in generality of the solution. Thus it is usually found necessary to resort to step-by-step trajectory calculations or computing machinery to investigate the influence of design variables in a particular problem. It may be observed, however, that the importance of ramjet climb and/or acceleration is closely related to the range objective and usually is of small proportion for very long-range missiles. For missiles with short-range, high-altitude objectives this phase can be the main problem. Under the latter condition, it is also obvious that lift-drag ratio ceases to be a meaningful parameter in the previous sense, although it may still play a part if trajectory maneuvers are required.

Some insight into the ramjet booster problem is given by Fig. 3.2-20. Variation of the launching weight factor with selected staging variables is shown. The primary variable,

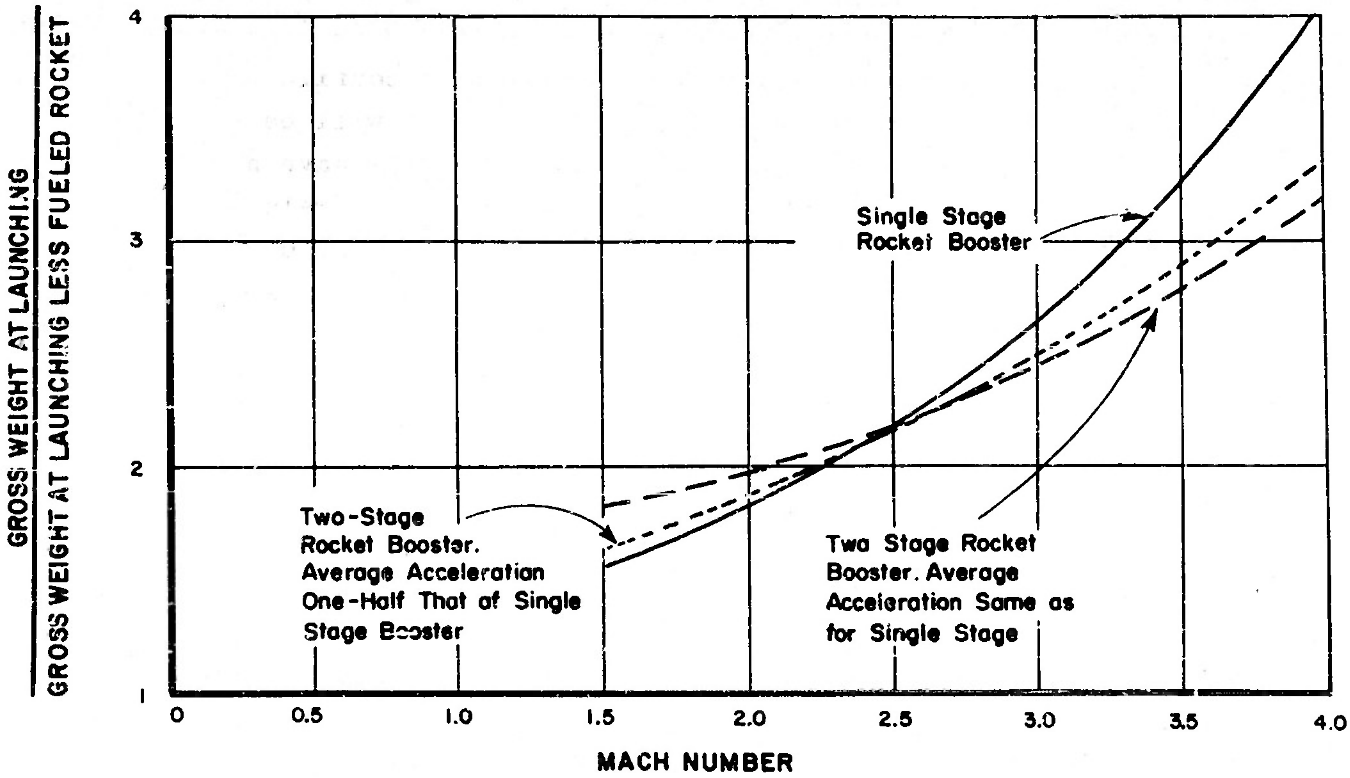

Fig. 3.2-20 THE EFFECT OF STAGING ROCKET BOOSTERS FOR RAMJET MISSILES AS A FUNCTION OF BOOST MACH NUMBER REQUIRED AND ACCELERATION PROGRAM

terminal Mach number required, has been chosen as consistent with ramjet operation characteristics. The base curve (from Fig. 3.2-19) demonstrates the relation of launching weight factor to terminal Mach number for a single rocket stage, constant average acceleration case. If it is assumed that four rockets make up the single stage, then the two-stage curve labelled "average acceleration one-half that of single-stage booster" may be interpreted as a boost condition in which rockets identical to the base case are operated successively in pairs to provide the final Mach number. For the remaining two-stage case, redesigned rockets with half the action time individually of the two preceding cases may be visualized operated successively in pairs to provide terminal Mach number in the same time interval as the single-stage booster. The results indicate that staging, under the conditions assumed, is not advantageous for a terminal Mach number requirement less than roughly M = 2.5. For higher terminal Mach number, staging becomes progressively more advantageous.

The effect of an acceleration program is of some interest. If launching conditions permit a reduced acceleration from the base assumption, then at M = 2.5 the reduced acceleration two-stage booster will provide appreciably higher starting altitude for the missile because of the longer booster action time. At slightly higher Mach number a smaller rocket stage weight for the base acceleration, two-stage case would require detailed evaluation to determine if the additional booster weight required for the reduced acceleration two-stage missile was compensated for by increased altitude. For still higher Mach number, it is possible for two-stage missile curves to join again due to rapidly increasing terminal boost altitude and consequent rapid lowering of the speed of sound at terminal altitude for the reduced acceleration case.

A more extensive treatment of the launching problem may be found in the chapter devoted to that subject (Chapter 17).

Terminal Flight

Terminal flight in many cases must be analyzed in a manner comparable to that used for the acceleration and climb phase. In addition to dependence on range, the importance of the final phase is determined by the path followed, to a greater extent than is usually the case for the initial phase, where a quick, steep, climb to altitude is usually indicated. Steep dives, for example, are very sparing of fuel. However, if the dive must be terminated by a return to approximately level flight for a short cruise period at low altitude, fuel consumption may be very high for reasons previously discussed.

It is obvious from the nature of the preceding remarks that parameter studies of important variables of initial and terminal phases of flight are likely to consume far greater relative effort to achieve satisfactory understanding than is the case for the cruise phase. In so far as possible, reliance must be placed on previous studies of similar nature to obtain first approximations. When such information is lacking, equations of limited usefulness incorporating restrictive assumptions may be derived to provide starting points for more detailed studies.

Engine Comparison for Long-Range Missiles

The objective of the previous discussion will have been met if the reader has developed an appreciation for missile and booster performance relationships. In the final analysis, it is necessary to form a composite aerodynamic and propulsion design. As in other activities, compromise and judgment inevitably temper the final result in ways not foreseen at the initiation of design. Nonetheless, it is possible to provide satisfactorily definitive answers in most cases in which results are not sensitive to the logarithmic nature of the range curve.

Figures 3.2-21 and 3.2-22 have been prepared to demonstrate the composite effect of Mach number and engine selection on weight of the missile and weight at launching. High acceleration rocket boosters have been assumed for all but one case to normalize the comparison. Under these conditions, for example, the reciprocating engine craft need not have a landing gear, nor is it necessary to provide a large engine for take-off and acceleration. Furthermore, a similar launcher may be presupposed for all but the excepted case. (An exception has been made for the variable velocity rocket; imposition of an initial acceleration constraint would increase the plotted weight by a substantial factor.) Certain rather clear-cut facts are evident from the figures. As would have been expected from previous paragraphs, the reciprocating engine is without peer in the low subsonic regime. Among air-breathing engines, the turbojet is superior in the high subsonic and transonic region while the ramjet dominates the supersonic regime above M = 2.0. The specific fuel consumption of the rocket is so high that this engine is ruled out for constant velocity, atmospheric flight paths, in spite of low engine weight and high thrust per unit cross-section area. If the velocity constraint is unnecessary, then high acceleration plus coast rockets are possible for this application, but are heavier in weight than ramjets when it is permissible to operate the ramjet near its optimum Mach number.

Emphasis has been placed previously on the logarithmic nature of the range curve. Figure 3.2-22 may be contrasted with Fig. 3.2-21 for a demonstration of the importance of this effect. For example, in the reciprocating engine case degradation of the lift-drag ratio by 20 per cent results in a weight increase of 100 per cent at M = 0.9. At M = 0.3, a similar lift-drag degradation results in a 12 per cent weight increase. Similar examples may be found for the other engines. It is obvious that constraints (e.g., combustion chamber pressure

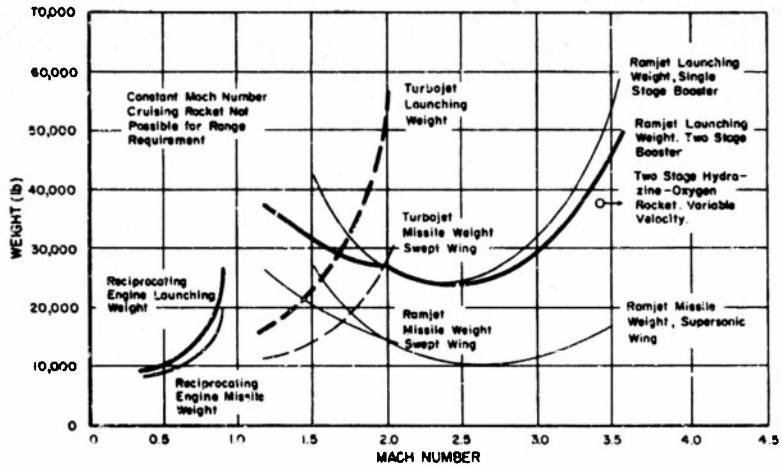

Fig. 3.2-21 MISSILE AND LAUNCHING WEIGHT AS A FUNCTION OF ENGINE TYPE AND MACH NUMBER FOR TYPICAL CONFIGURATIONS. RANGE 2500 NAUTICAL MILES. PAYLOAD 4000 POUNDS.

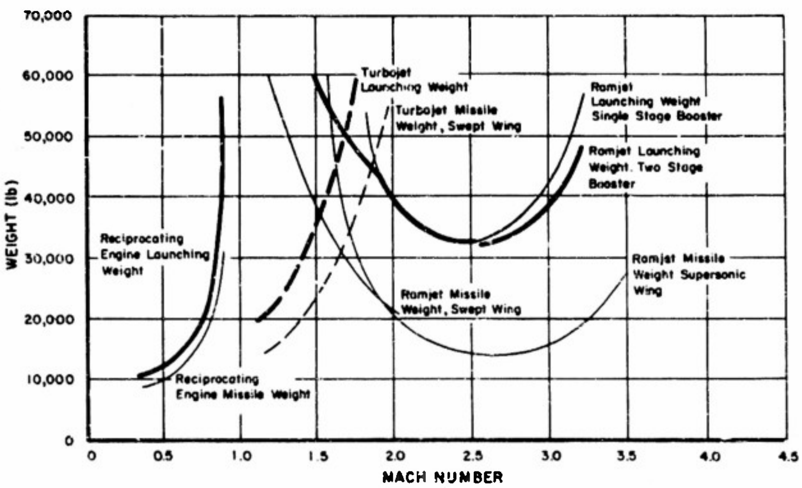

Fig. 3.2-22 MISSILE AND LAUNCHING WEIGHT AS A FUNCTION OF ENGINE TYPE AND MACH NUMBER FOR TYPICAL CONFIGURATIONS. RANGE 2500 NAUTICAL MILES. PAYLOAD 4000 POUNDS. (LIFT-DRAG RATIO OF FIG. 3.2-21 DEGRADED 20%)

requirements) or small errors in judgment can have an enormous effect on comparisons based on operation near the maximum capability of a particular airframe and propulsion system (i.e., in the "knee" of the Breguet curve). Comparisons made under such conditions must be taken with a grain of salt.

Certain of the above considerations disappear in the usual comparison based on the designer's judgment as to representative values of flight parameters. A valid comparison must, however, admit the treacherous nature of calculations made for an extreme range condition. The wise designer will seek systems of minimum sensitivity to this condition and, when possible, avoid specification of undesirable combinations of airframe and propulsion unit and flight conditions.

Criteria for Engine Selection and Missile Optimization

It is meaningless to discuss engine selection and optimization in a context of weight and/or size without consideration of the "system" of which the missile is but a part. Meaningful comparisons must be based on an evaluation of how individual items may be fitted and tailored to assure "optimum" operation of the whole in meeting its fundamental objective. It follows, of course, that a treatment of this nature will not always provide the same standard for judgment of merit. The shipboard antiaircraft missile defense problem will be used as an example.

Antiaircraft missile naval defense planning is complicated by the existence of a large and expensive fleet of ships. Economics dictate that these hulls must be utilized. Thus a structure developed for high density storage must be adapted to a low density item. Various arbitrary limitations must in turn be accepted by the missile system designer. It is obvious also that restrictive thinking of this type must be considered in a framework of four-dimensional (i.e., including time) warfare lest the ultimate be neglected and sacrificed in the narrowness of the problems of the moment. It is in this general context that the following remarks must be considered.

Shipboard handling studies have in the past consistently demonstrated that criteria of missile excellence (and propulsion selection and optimization) must be related to the type of handling mechanism and size and type of ship under consideration. Obviously, there is also an interrelationship between the machinery required (even though various alternates are possible) and missile-booster characteristics. An example of the effect of ship choice may be visualized by considering a vertical handling system on a small ship which can provide but two deck heights for missile operations. It is certain in this case that missile and booster of contemporary antiaircraft missiles must be separated until just prior to firing. On the other hand, a larger ship may provide sufficient clear depth to make possible stowage of mated missile and booster. Machinery requirements will certainly be less severe for the second case. If it is assumed that aerodynamic surfaces are to be attached just before firing in both of the above systems, then the missile stowage criterion will be as follows. For the separated missiles and boosters, length of the longest component, and missile plus booster diameter, will be significant. For the mated missile and booster, over-all length of the combination and diameter of the largest component will be critical. Certain over-riding limitations exist. For example, the booster diameter should not be designed to a lesser value than the missile diameter. If booster diameter is the lesser, transfer of mated missile and booster by rail is rendered difficult. In general, this latter objection can be met only by exaggerated height launching and handling lugs, which are objectionable in themselves. Separation difficulties may also be anticipated if tandem boosters are smaller than the missile. The constraints thus applied should guide the airframe and propulsion designer to proportion missile and booster to achieve maximum target handling capacity. It is worthy of note that no reference to weight of components appears in the above discussion. It is generally true for shipboard missiles that weight plays a secondary role. This is due to the low density of the missiles and associated handling equipment as compared to the removed gun turrets. A carte blanche does not exist, however, for weight because of the possible effect of such weight in denying inclusion of other new defensive or offensive equipment.

The effect of missile plus booster length on the silhouette of a ship can be significant. It is an unfortunate fact that many two-stage missiles (missile plus booster) have a length sufficiently great to create an adverse effect on adjacent launcher clearance, and ultimately on the balance of the ship itself through secondary effects on guidance and search radar placement. Complete elimination of firing interference does not appear possible even with appreciably shorter missiles. Many feet must be saved to be effective. While the wrap-around booster is effective in this role, the disadvantages from a below-decks stowage point of view are so drastic as to rule out this alternative. For short-range applications, a single-stage missile which integrates the sustainer and booster, can be effective under proper conditions. Previous proposals for such integration have required liquid rocket propellants for the boost phase. The latter appear so hazardous and the problems of handling so formidable aboard ship, that almost any alternative involving solid rocket propellants seems preferable. It is possible that new developments in solid fuel ramjets may provide the key to a satisfactory solution.

Limiting lengths exist due to the compartmentation necessary to insure security under partially flooded conditions.

In the horizontal plane the distance between water-tight bulkheads (approximately 32 feet) is critical. In the vertical plane deck heights (approximately 8 feet) are limiting. In the latter case, vertical integration of compartments tends to result in limiting lengths which are integers of eight feet. In general, specification of a missile and/or booster length which fails to make full use of the committed ship's length (e.g., three deck heights for a large ship) utilizes volume inefficiently.

The concept of deck height increments can be applied to storage of wing surfaces as well as missiles and boosters. In vertical loading systems, it is desirable to limit the maximum chord dimension of surfaces to a value somewhat less than the expected maximum clear height between decks. Since this value is of the order of 7-1/2 feet, the maximum chord dimension should not exceed approximately 80 inches. Obviously, if two or more surfaces are stored, the sum of chords (and/or span as an alternative if panels are very small) should not exceed this value, less an allowance for partitions. To minimize span and airfoil thickness, full advantage should be taken of this dimension when to do so is not inconsistent with other objectives. The fullest use of panel stowage area can be achieved in this manner.

The earliest proposals for shipboard guided missile handling systems carried disassembly of components to the point where sub-sections of the body (warhead, guidance, sustainer, etc.) were assembled, missile and booster mated, and surfaces attached prior to transfer to the launcher. With the passing of time it has become increasingly apparent that a practical system fast enough to keep pace with possible tactical situations must be founded on a minimum number of assembly operations aboard ship. Presently planned guided missile ships will use missile and booster assembled as a unit in a ready ring. Twelve wing, tail, and booster fin panels must be attached prior to firing. To insure a minimum assembly time, each panel must be fitted simultaneously, thus requiring twelve men per missile. Interpreted in terms of a full missile ship conversion, this means that 96 men are required for assembly of surfaces. It is obvious that reduction of the number of men required would be most advantageous (battle station requirements fix wartime complement). This is particularly true when it is recognized that approximately three tons of supporting equipment, stores, etc., are required for each man's maintenance aboard ship. While this problem does not seem related to propulsion upon first thought, reflection will serve to establish that an integrated propulsion system (i.e., booster and sustainer) would permit a reduction of surfaces from twelve to possibly three or four (even two in principle). The saving in manpower and supplies which would result, would make available for other assignment a substantial number of men and increase the ship's volume which could be devoted to guided missiles. Interactions which involve guidance, manufacturing facilities, cost, etc., also must be included in the complete evaluation of a guided missile system.

Naval use of other than antiaircraft missiles must be assumed. Offensive use of atomic warhead missiles, for example, will call into being not only considerations similar to those previously discussed, but factors related to delivery tactics and strategic planning. Dimensional characteristics of missiles will be critical to a higher degree for submarine delivery than for surface ship delivery; however, the advisability of using surface vessels in this task can be seriously questioned. Delivery by a surface ship, in fact, implies a task force operation for adequate ship protection, whereas single submarines can operate effectively, cheaply, and with a maximum surprise element in their favor. Use of submarines will place certain restrictions on propulsion of a fundamental nature. Possibly the most important will be related to a demand for high acceleration during launch to minimize launcher problems. Other restrictions will be recognized by the reader.

It quite evident from the foregoing that optimization and comparison studies can be quite superficial unless a broad viewpoint is maintained. The designer of antiaircraft missiles must coordinate his work closely with the activities of the air defense planner to insure that adequate defense is achieved without undue compromise of offensive power. The designer of bombardment missiles must work in a further enlarged context, embracing tactical and strategic planning. The challenge presented is one requiring the efforts of many. It must be accepted to insure the national welfare.

NOMENCLATURE

Symbol	Definition	Units
A	reference area	ft^2
a	speed of sound	ft/sec
C_D	drag coefficient	
C_L	lift coefficient	
D	drag	lb
F	thrust	lb
g	acceleration	ft/sec^2
I_f	specific impulse	lb thrust sec/lb fuel
L	lift	lb
M	Mach number	
q	dynamic pressure	lb/ft^2
R	range	ft
t	time	sec
U	velocity	ft/sec
W	weight	lb

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